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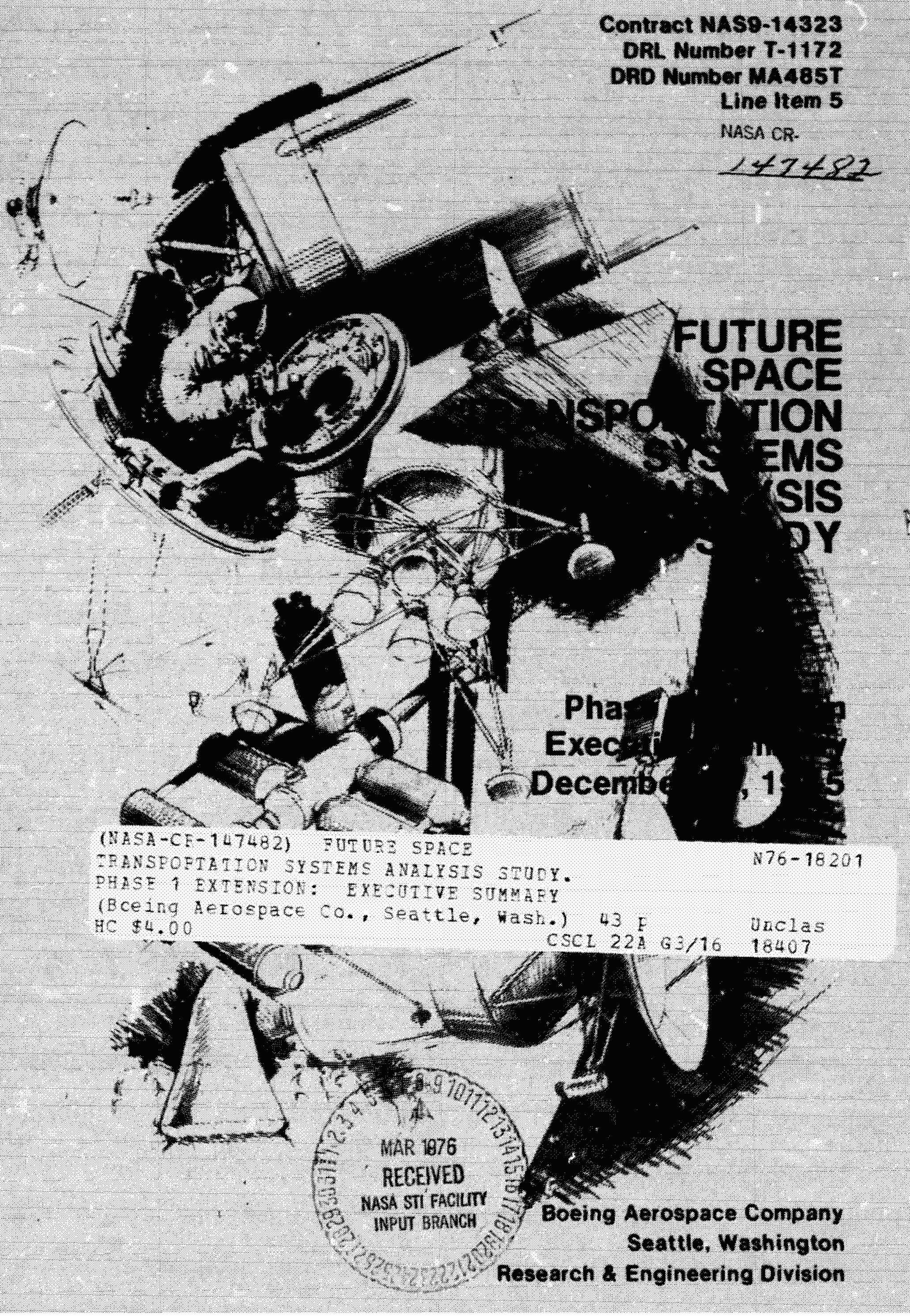

FUTURE SPACE TRANSPORTATION SYSTEMS ANALYSIS STUDY

Phase 1 Extension
Executive Summary
December [illegible], 1975

Boeing Aerospace Company
Seattle, Washington
Research & Engineering Division

FOREWORD

The Future Space Transportation System Analysis Study, NASA Contract NAS9-14323, is managed by the NASA Lyndon B. Johnson Space Center (JSC) and is being performed by the Research and Engineering Division of The Boeing Aerospace Company in Seattle, Washington. The Contracting Officer's Representative (COR) is Harle L. Vogel of the Future Programs Division of JSC. Study management team members assisting the COR are:

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This document is the executive summary report at the completion of the Phase I extension (December 19, 1975). It summarizes results of the study up to the present time. Phase II, now in progress, will be complete at the end of December, 1976. Requests for information should be directed to Gordon R. Woodcock of the Boeing Aerospace Company in Seattle or Harle L. Vogel of the Future Programs Division of the Johnson Space Center in Houston.

1.0 INTRODUCTION

The Future Space Transportation Systems Analysis study is an analysis of potential future space programs beyond the scope of the current shuttle traffic model, intended to determine their transportation needs, and an evaluation of alternative ways of evolving future space transportation systems from the baseline Space Transportation System (space shuttle and upper stage), to meet those needs. Objectives for the entire study are indicated in Table 1.

Table 1. Study Objectives

OBJECTIVE	STUDY PHASE
• DEFINE POTENTIAL TRANSPORTATION REQUIREMENTS RESULTING FROM A RANGE OF POTENTIAL FUTURE SPACE MISSION OPTIONS	I
• IDENTIFY POTENTIAL MISSION/ TRANSPORTATION MODES AND SYSTEM OPTIONS CAPABLE OF SATISFYING THE REQUIREMENTS	I
• DEFINE THESE MISSION/TRANSPORTATION MODES AND SYSTEMS • VEHICLE CHARACTERISTICS • PERFORMANCE • COST	I Extension
• SYNTHESIZE AND SELECT, FROM THESE DEFINED OPTIONS, OPTIMIZED TRANSPORTATION SYSTEMS FOR A RANGE OF POTENTIAL SPACE PROGRAM EVOLUTIONS	II
• DEFINE ADAPTIVE EVOLUTIONARY STRATEGIES FOR FUTURE SPACE TRANSPORTATION CAPABILITY DEVELOPMENT	II

The sequence of analysis steps for the Future Space Transportation Systems Analysis Study is shown in Figure 1. Phase I (complete) set the stage for the subsequent phases by identifying and defining missions and potential transportation modes and options. The Phase I extension developed the data base for the transportation modes needed to accomplish Phase II. Through the Phase I Extension each mission option was considered independently of the others; transportation modes

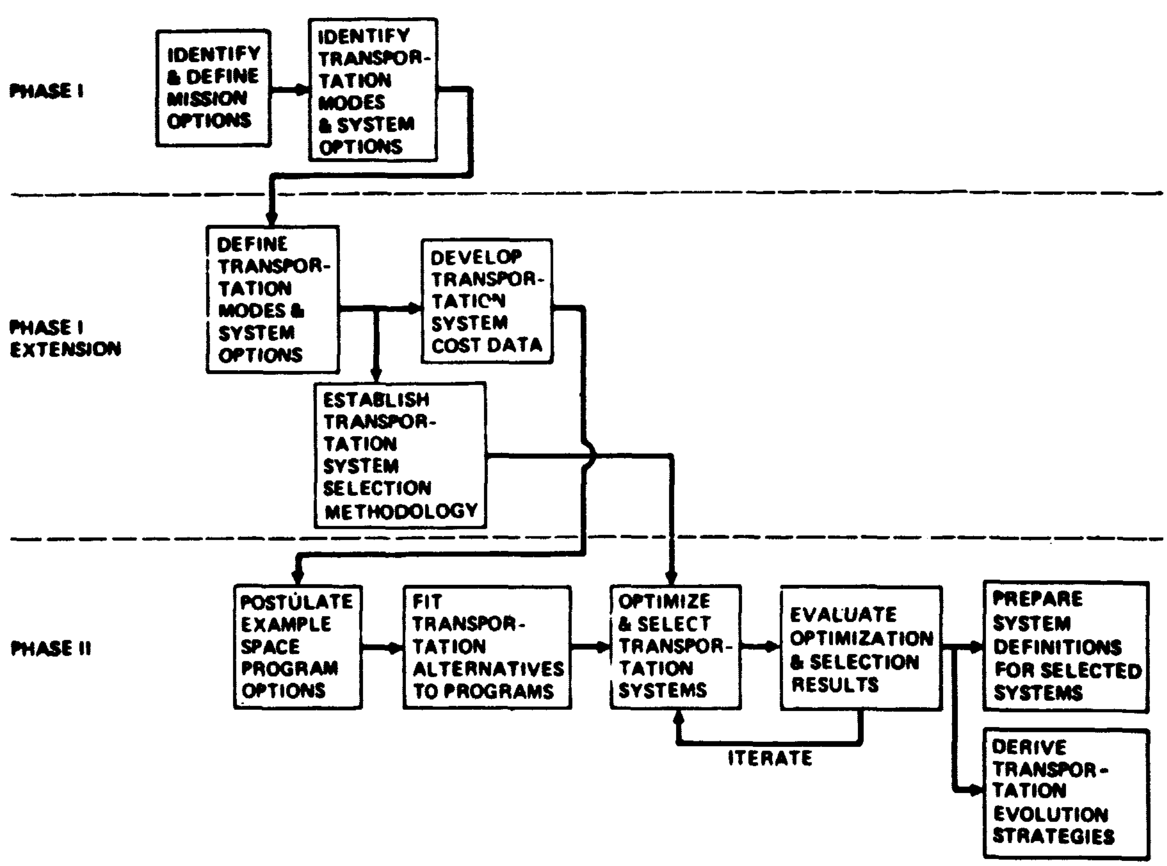

Figure 1. Overall Study Logic

and systems were tailored to each mission option without consideration of potential alternate uses.

In Phase II, the individual mission options are being combined in a variety of ways to form potential integrated programs, and the transportation alternatives will be adjusted as needed to encompass the alternative uses within each integrated program. A broadly-scoped system optimization and selection analysis will define the best overall transportation system approaches and evolution strategies.

Current schedule status and plans are shown in Figure 2. Timing of the Phase II schedule has been adjusted to mesh with data availability from the Heavy Lift Launch Vehicle (HLLV) study.

The FSTSA study has technical interfaces with several other studies. The more significant ones are shown in Figure 3. Interchanges with the HLLV and space based power studies occur on a continuing real-time basis; with the others, somewhat less frequently.

2.0 PROGRAM AND MISSION OPTIONS

The study began by identification of program and mission options and determination of their transportation requirements in terms of payload sizes, masses, and delivery and returns requirements. As the study has progressed, new mission options have been added within the program areas. The set currently under consideration is shown in Table 2; descriptions of each program area are presented on the next several pages.

2.1 LOW EARTH ORBIT SPACE STATION

Three options for manned stations in a low Earth orbit have been considered. The modular space station illustrated in Figure 4 can be delivered to orbit in modules by the Space Shuttle and assembled in orbit. As such, it does not require advanced space transportation. A unitary space station consists of a single large core module with attached application and science modules (ASM's) tailored to specific missions. The unitary station is too large for the Shuttle and must be placed in orbit by a

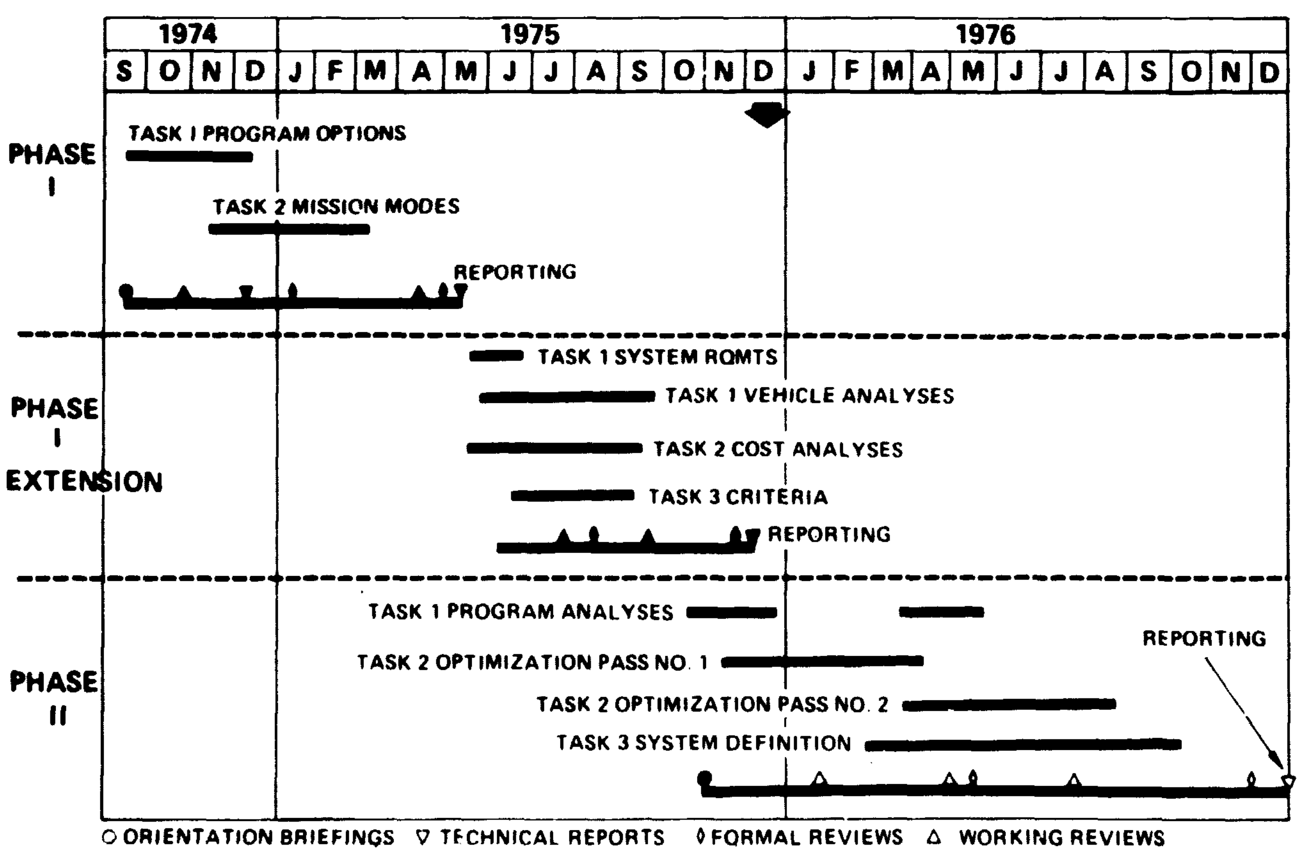

Figure 2. Future Space Transportation Systems Analysis Schedule Overview

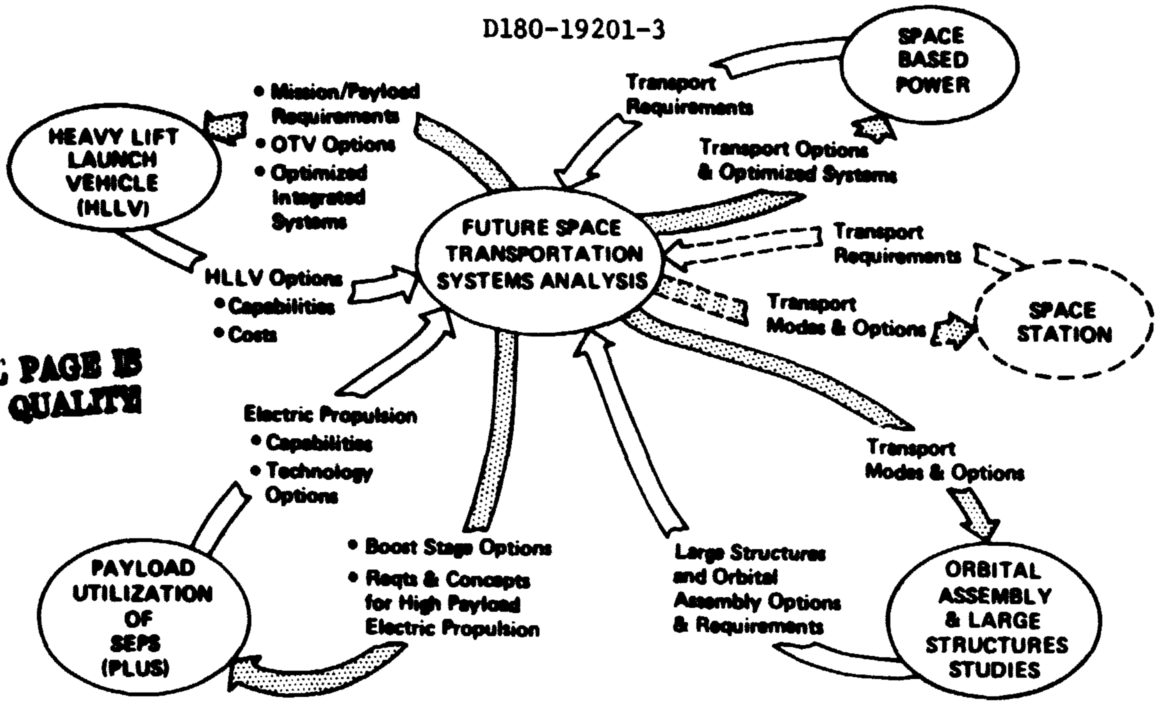

Figure 3. Study Interfaces

Table 2. Program Options

Program	Missions	Objectives
1. Low Earth Orbit Space Stations	• 12-man modular or unitary station • 60-man space base	• Assembly operations for large structures • Broad spectrum earth observatory • Develop space manufacturing • Scientific investigations
2. Geosynchronous Operations	• Satellite maintenance sortie • 8-man modular or unitary station • 50-man station	• Maintenance and repair of automated spacecraft • Earth observations • Communication/navigation • Maintenance base for solar power stations
3. Independent Lunar Surface Sorties	• 4-man self supporting landing	• In-depth exploration of selected areas
4. Orbiting Lunar Station	• 8-man modular or unitary station with surface sortie	• Broad spectrum surface observation • 4-man, 28-day sorties
5. Lunar Surface Base	• 6-man, 6-month • 12-man, semipermanent • 200-man, semipermanent	• Astronomical observations • Surface exploration • Indigenous material utilization
6. Manned Planetary	• Manned Mars landing • Opposition • Conjunction • Venus swing-by	• 3-man, 30 day sortie • Planetology • Effects of modifying forces • Search for life forms
7. Automated Lunar Exploration	• Orbital observatory • Backside lander • Relay satellite	• Broadband scientific observation • Long duration Rover with sample return
8. Automated Planetary Exploration	• Mars lander • Jupiter atm probe • Ganymede lander	• Rover/sample return • Invest upper cloud system • Orbital observation and surface sample analysis
9. Nuclear Waste Disposal	• Refined waste • Total waste	• Permanent waste disposal
10. Satellite Energy Systems	• On-orbit power generation • On-orbit power reflectors • Pilot/demonstration programs	• Commercial electric power • Long range power transmission

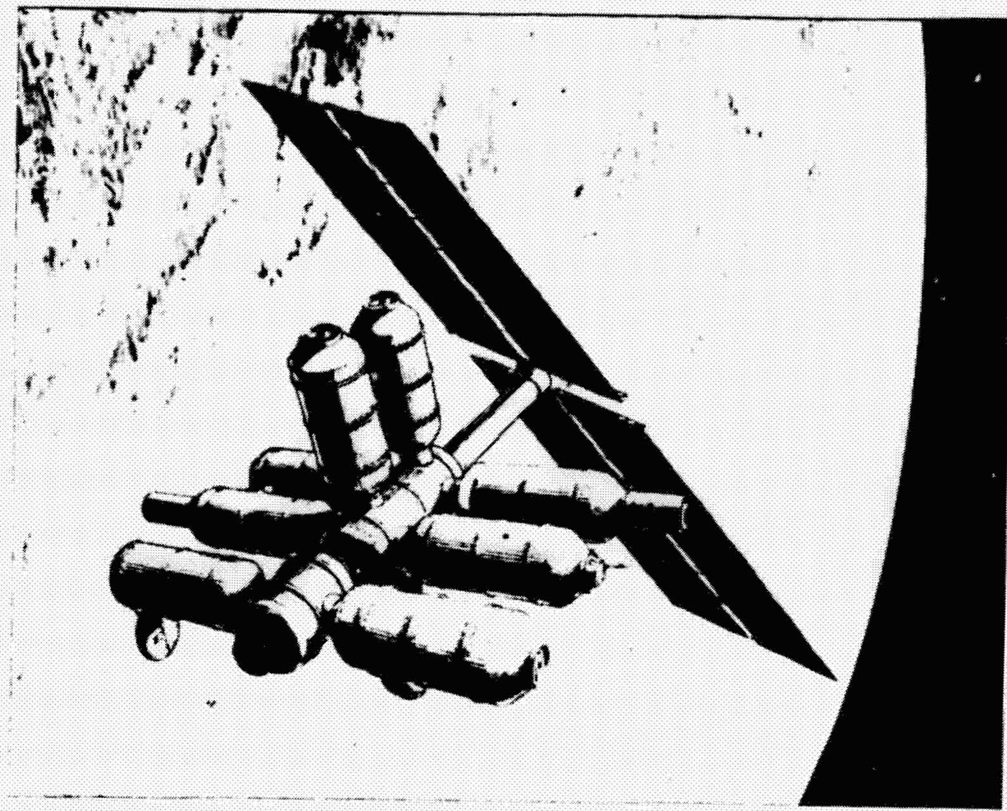

Figure 4. Low Earth Orbit Space Stations

heavy lift vehicle. Both of these stations normally carry a 12-man crew. The third option is termed space base. The base is built up from the large unitary station size modules in a manner analogous to the assembly of smaller modules to form a 12-man modular station. The space base will support a crew of up to sixty men and can be expanded beyond that if desired. These stations nominally operate at approximately 500 km (270 n.mi.) altitude and either 28.5, 55, or 97 (sun sync.) degrees inclination.

The objectives of these stations are as follows:

- Develop technology and operational capability for orbital assembly of large systems and for space manufacturing.
- Provide long term manned space residence.
- Conduct scientific investigations of the near Earth space.
- Provide a broad spectrum Earth observatory.
- Demonstrate adaptability to observation of other celestial objects (solar, stellar, etc.).

The mission system elements for the 12-man modular station include the basic station modules, application and science modules (ASM), crew transfer modules (CTM), and resupply modules (RM). Nine station modules are required to provide quarters for the 12-man crew, supporting subsystems, laboratories and consumables. A solar array/battery system provides electrical power of 25 kw. The unitary space station provides a single large module housing the crew, general purpose labs and

subsystems. Crew transfer modules, resupply modules, and application and science modules have the general functions as defined for the modular station. Crew rotation and resupply interval for the 12-man stations is three months, assuming use of the space shuttle for this function.

The space base consists of five large modules that each accommodate 12 crewmen and lab facilities, a base subsystem/hub module and two nuclear reactor modules that provide electrical power.

Crew rotation and resupply functions are performed in the same general manner as for the 12-man stations; resupply interval assuming use of the shuttle is approximately 3 weeks rather than 3 months.

The space station options depicted and described were obtained from previous NASA studies of science-oriented stations. It is expected that stations designed primarily to develop and support space assembly operations will be different in appearance but that their transportation requirements will not be markedly different.

2.2 GEOSYNCHRONOUS OPERATIONS

2.2.1 Geosynchronous Space Stations

Geosynchronous orbit is the preferred location for a number of science and applications space operations. Examples of these operations include weather reconnaissance, communications/navigation, global environmental science, Earth survey, and the generation or transmission of electrical power for terrestrial needs by satellite energy systems. Conduct of these operations may be accomplished through use of manned space stations and automated spacecraft.

The missions selected to represent advanced geosynchronous transportation requirements were manned space stations and a manned sortie to service automated spacecraft. Satellite energy systems were treated as a distinct program option.

Figure 5 shows delivery of the second half of an 8-man modular geosynchronous station by the upper stage of an orbit transfer system at the time of rendezvous with the first half of the station.

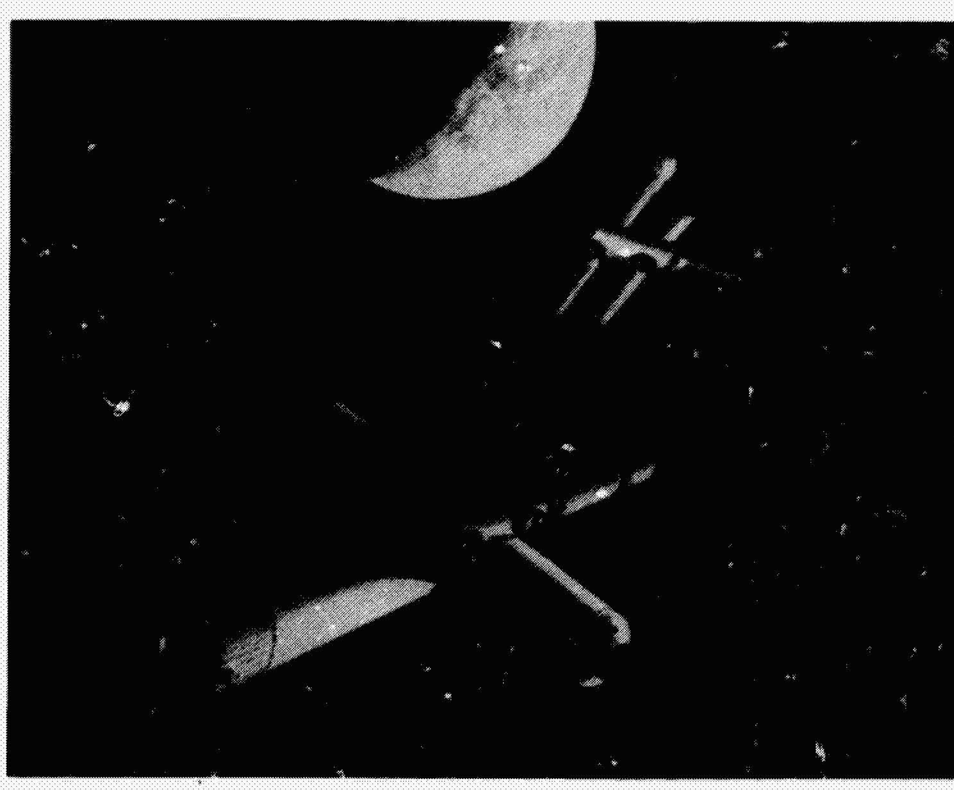

Figure 5. Geosynchronous Space Stations

Delivery of the station in two flights is compatible with sizing of the transportation system for the recurring requirement of crew rotation and resupply.

The reference geosynchronous space station (GSS) mission consists of a modular station that can be continuously occupied by a crew of eight and can accommodate both Earth application and science sensors. System elements making up this concept include the basic station modules, application and science modules, crew transfer vehicle and resupply modules.

Nine station modules provide quarters for the eight-man crew, supporting subsystems and consumables. The functions provided by these modules are as follows: two core modules house basic station subsystems and the docking provisions for all the other modules; two modules each provide crew quarters for four men and eight in an emergency; two modules serve as command/control centers with one also providing the radiation shelter; one module provides the electrical power system; one module is used for the galley and recreation purposes; and the final module houses cryogenics and provides storage.

A unitary station option for this mission is also described in the study technical report. The eight-man station options require crew rotation and resupply at six-month intervals. Delivery and return payloads are 24 950 kg (55,000 lb) and 14 970 kg (33,000 lb) respectively. Total masses of the modular and unitary stations as initially delivered are 125 000 kg (275,000 lb) and 86 300 kg (190,000 lb) respectively.

A brief study was made of transportation requirements for a 50-man geosynchronous station. The selected crew rotation and resupply interval was 2 months with delivery and return payloads of 40 100 kg (88,400 lb) and 23 100 kg (50,900 lb) respectively. The 50-man station delivery mass was 423 000 kg (931,000 lb). Station deliveries were delivered with station element sizes compatible with orbit-to-orbit transportation systems tailored to crew rotation and resupply requirements.

2.2.2 Geosynchronous Satellite Maintenance Sortie

This mission was selected as an example of geosynchronous operations on a more modest scale. Current estimates of the quantity of geosynchronous automated satellites range from 180 to over 400 by 1990. Economics associated with operating the satellites will probably necessitate repair and refurbishment rather than disposal when a failure occurs. Complexity of the satellite however, may prevent maintenance by automated vehicles. As a result of the above factors, the need for manned sorties to repair and refurbish automated geosynchronous satellites is a probable requirement.

The reference geosynchronous satellite maintenance sortie (GSMS) mission consists of a four man crew performing one week of maintenance operations, visiting four satellites with transfers up to 15° longitude between each visit. Figure 6 shows the GSMS vehicle approaching a large infrared telescope.

The major payload elements in this mission are a crew transfer vehicle (CTV) to house the four man crew for one week and the necessary repair and refurbishment provisions. The total initial payload is 6570 kg (14,485 lb) and returned payload is 5970 kg (13,160 lb). Initial allocation for repair and refurbishment provisions is 1000 kg (2200 lb).

2.3 INDEPENDENT LUNAR SURFACE SORTIES

The independent lunar surface mission has as its objective the local exploration of selected lunar areas. The term "Independent" signifies that each mission is self-supporting as were the Apollo lunar missions. Logistics flights or support missions are not required to return the mission crew to Earth.

Each mission leaves Earth orbit, transfers to the moon, enters a lunar orbit, lands four men and 4500 kg (10,000 lbs) of mission equipment on the lunar surface for a 14-day exploration stay, and then returns to Earth.

A representative independent lunar sortie configuration is illustrated in Figure 7. The concept consists of a crew and equipment module, lunar transport vehicle for landing and take-off and two experiment/exploration payloads. The lunar transport vehicle illustrated consists of a single stage that uses LO_2/LH_2 propellants and is used for descent from lunar orbit, landing and the ascent back to lunar orbit. Single-stage and 1 1/2-stage

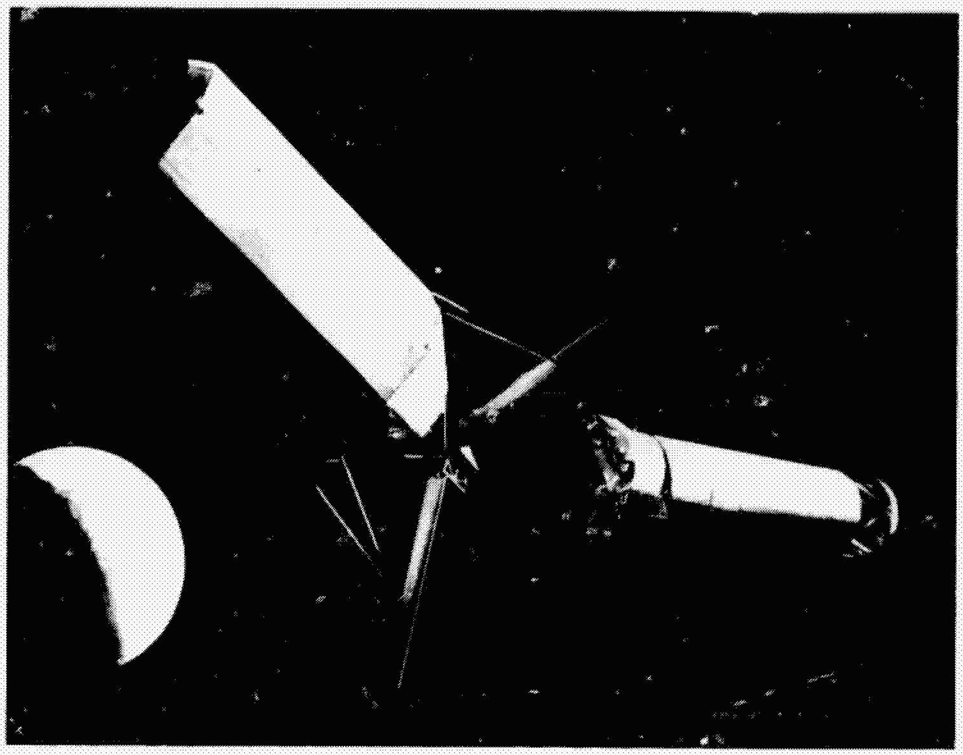

Figure 6. Geosynchronous Satellite Maintenance Sorties

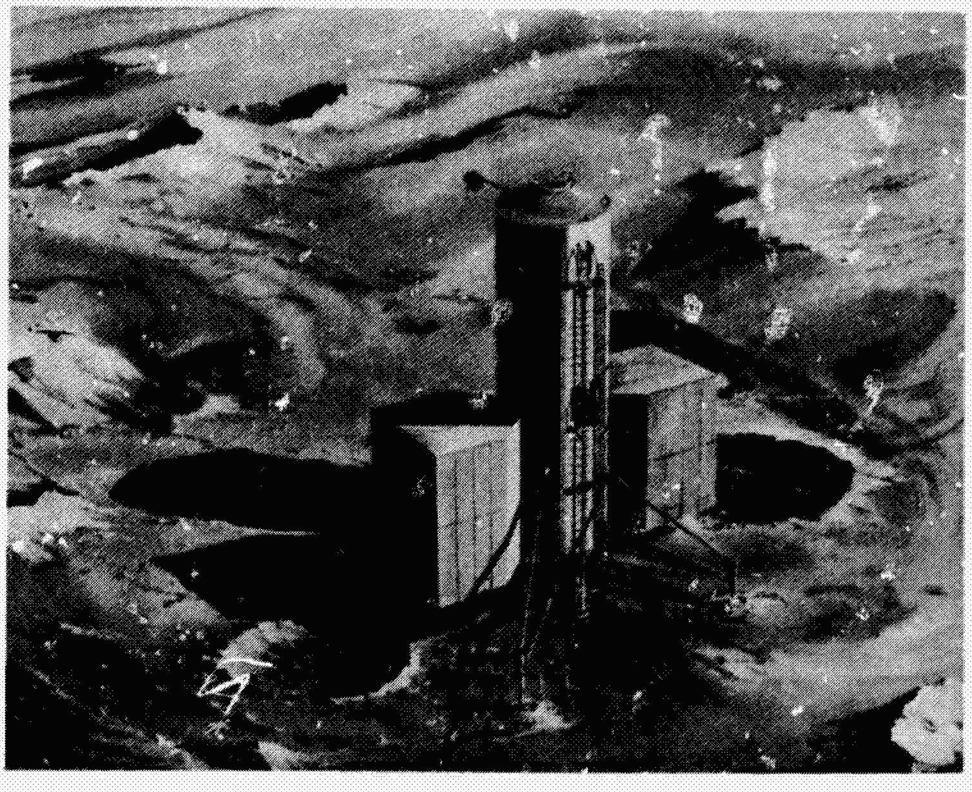

Figure 7. Independent Lunar Surface Sorties

LTV's using LO_2/LH_2 and LO_2/MMH propellants were investigated. On the ILSS mission the LTV is expended after the crew and equipment module is transferred to the orbit transfer vehicle for return to Earth.

2.4 ORBITING LUNAR STATION PROGRAM

An extensive investigation of the moon involving orbital science and surface exploration of a number of sites for up to 30 days could employ an orbiting lunar station. The objectives of this program would be as follows:

- Perform a broad spectrum observation of the lunar surface.
- Conduct surface sorties.
- Support and control unmanned orbital and surface operations.

The systems elements associated with the orbiting lunar station (OLS) include a space station, lunar transport vehicle (LTV), resupply modules (RM), fluid module (FM) and crew transfer vehicles (CTV).

The flight configuration for a modular station is shown in Figure 8. Nine modules are required to provide the required volume for a crew of eight, subsystems, and consumables. A tenth module contains science equipment and sensors. A unitary OLS could also be employed and would require only one habitat module rather than nine. The unitary option is described in the study technical report.

Two LTV's each provide capability to conduct a 4-man 28-day surface exploration. Landing and ascent payloads are 15 900 kg (35,000 lbs) and 11 400 kg (25,100 lbs) respectively. Exploration payloads include a lunar vehicle (LRV) and lunar flying vehicle (LFV). The LTV's also serve as emergency vehicles to transport the OLS crew back to Earth orbit should the OLS require evacuation or to rescue a crew stranded on the lunar surface.

A combination crew rotation/resupply flight occurs at 109 day intervals. Typical delivery and return payloads are 59 940 kg (132,000 lbs) and 6120 kg (13,500 lbs). Crew rotation is accomplished through use of a crew transfer vehicle (CTV). The CTV is sized to provide quarters for up to 8 crewmen during transits between Earth and

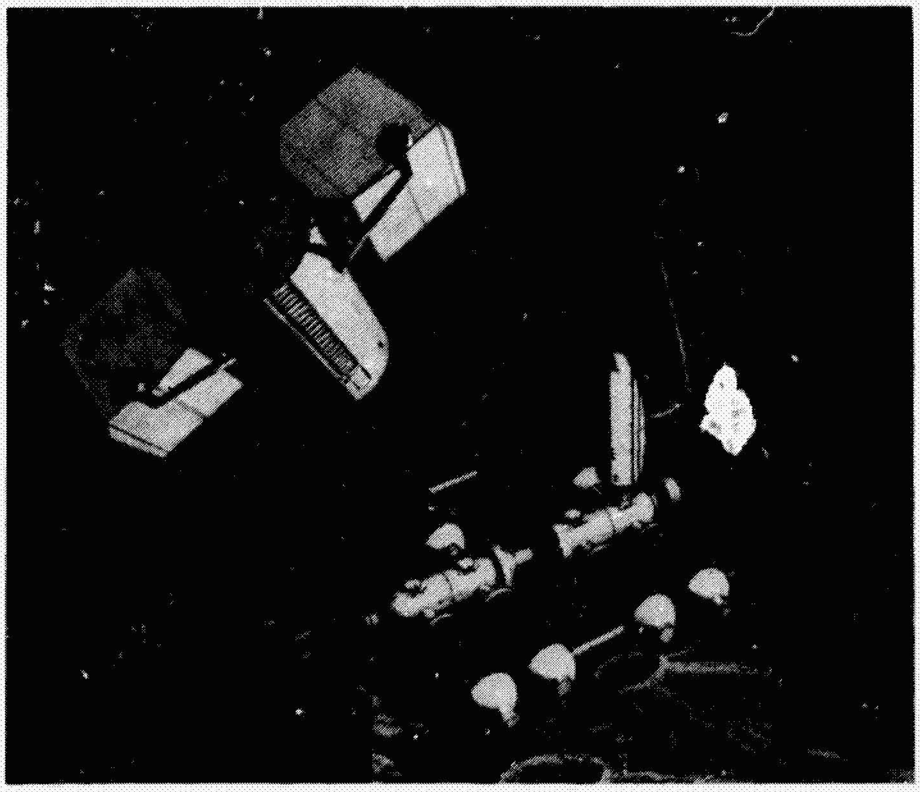

Figure 8. Orbiting Lunar Station Program

lunar orbit. The resupply module (RM) is a pressurized container that includes bulk cargo (e.g., food, clothes, etc.) for both OLS and LTV. The module is sized for a basic resupply interval of 109 days plus 55 days for contingency. The fluid module (FM) provides propellant to completely replenish one LTV and all lunar mobility vehicles and cryogenics for the OLS atmosphere.

2.5 LUNAR SURFACE BASE

Three representative lunar surface base (LSB) missions were investigated: 1) a 6-man temporary base with durations up to six months; 2) 12-man semi-permanent or permanent base as illustrated, with mission durations up to five years; 3) a 200-man semi-permanent base. Emphasis in transportation requirements description was given to the 12-man base, shown in the construction phase in Figure 9.

The objectives of the 12-man LSB were to conduct an evolutionary program leading to nearly permanent manned presence, while performing astronomical observations, deep drilling, remote explorations, local science and experiments on extraterrestrial resources utilization.

Accomplishment of this mission requires a variety of surface equipment, capability to rotate the crew, resupply capability, and transportation elements to move the equipment between the Earth and the moon.

An additional consideration for LSB missions involves recent studies investigating the feasibility of establishing space colonies or space industries. Some of these studies have assumed extensive use of lunar surface materials. The concept of colonies in space might lead the 12-man LSB mission to place more emphasis on pilot plant operations

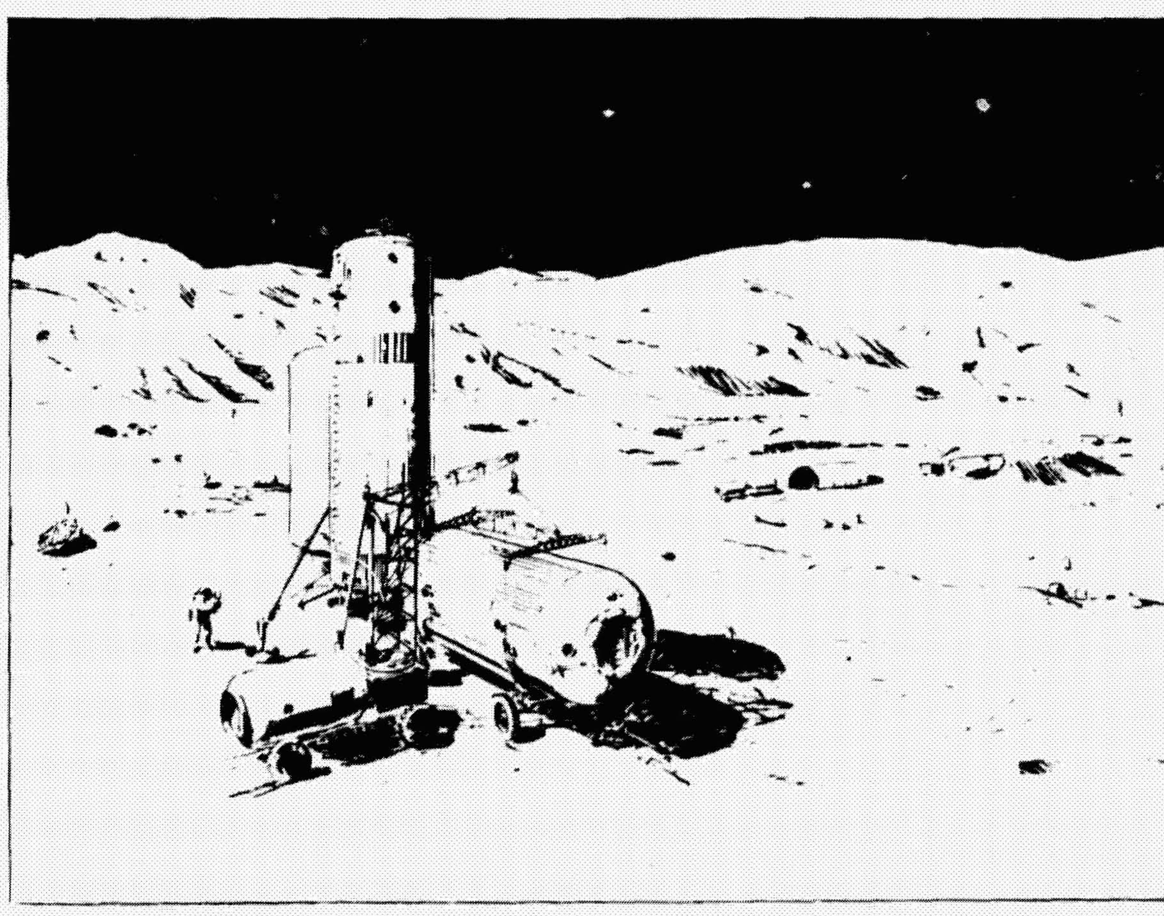

Figure 9. Lunar Surface Base

Table 3. Lunar Surface Base Payloads

BASE	TOTAL LANDED TO ESTABLISH BASE	CREW ROTATION & RESUPPLY	CREW ROTATION & RESUPPLY INTERVAL
6-man 6-month	50 000 kg (110,000 lbs)	not required	not required
12-man	145 000 kg (315,000 lbs)	27 700 kg (61,000 lbs) delivery 6 400 kg (14,000 lbs) return	164 days (5.5 months)
200-man	3.5×10^6 kg (7.7×10^6 lbs)	42 000 kg (92,600 lbs) delivery 10 000 kg (22,000 lbs) return	1 month

processing indigenous materials. Task 2 of Phase 1 was conducted using the original 12-man LSB definition. It is believed that the postulated pilot plant payloads are comparable to the science payload definitions used. A brief study of a 200-man base was conducted in order to scope the magnitude of this transportation requirement. A summary of payload masses is given in Table 3.

2.6 MANNED PLANETARY EXPLORATION

Manned Mars landing is the representative mission for the manned planetary exploration program. The objectives of this mission are to conduct an in-depth science program dealing with: Mars planetology, effects of modifying forces, composition, environment and possible life forms.

The major system elements associated with a manned landing on Mars include a mission module (MM), Mars excursion module (MEM) and Earth entry module (EEM). These elements along with unmanned probes form the mission spacecraft.

The mission module crew compartment provides the six man crew with a shirt-sleeve environment, quarters for living functions, operations center, experiment laboratories, radiation shelter and many of the subsystems required to support the above functions. This compartment is occupied by crewmen for the entire mission, except during the time when three crewmen descend to the Martian surface and during the Earth entry phase of the mission.

The MEM is used to land a three man crew on the surface, provide crew quarters and operations center for 30 days and return the crew to the mission spacecraft.

The Earth entry module (EEM) configuration is a six-man blunted biconic shape. The EEM systems are designed for 1 day's occupancy prior to Earth entry. The heat shield is designed by the highest Earth entry velocity expected from the opposition mission. The unmanned probes are used to check out the potential landing sites for the MEM, collect Mars orbital science data and explore the moons orbiting Mars.

Typical payloads are, delivery to Mars orbit 110 000 kg (242,000 lbs), depart Mars orbit 50 000 kg (110,000 lbs). The difference is primarily due to the MEM expended at Mars. The EEM mass is approximately 7900 kg (17,400 lbs).

Analysis of manned planetary exploration was limited to that necessary to determine the ability of propulsion vehicles defined for other missions to perform this mission.

2.7 AUTOMATED LUNAR OPERATIONS

The program objective is unmanned exploration of the lunar surface, including high latitude and backside regions, with the following capabilities:

- Surface mobility (long duration traverses)
- Deployed science stations (extension of ALSEPS net)
- Sample return (of material collected on traverses)
- Broadband scientific observation of the total lunar surface.

Shown in Figure 10 are general operational features of the various program hardware elements. The science satellite is located in polar orbit; it releases a subsatellite into the same orbit. A backside landing and a traverse by a deployed rover are also shown. Communications during backside operations are relayed by a "halo" orbit satellite. Science stations are set out by the rover, which

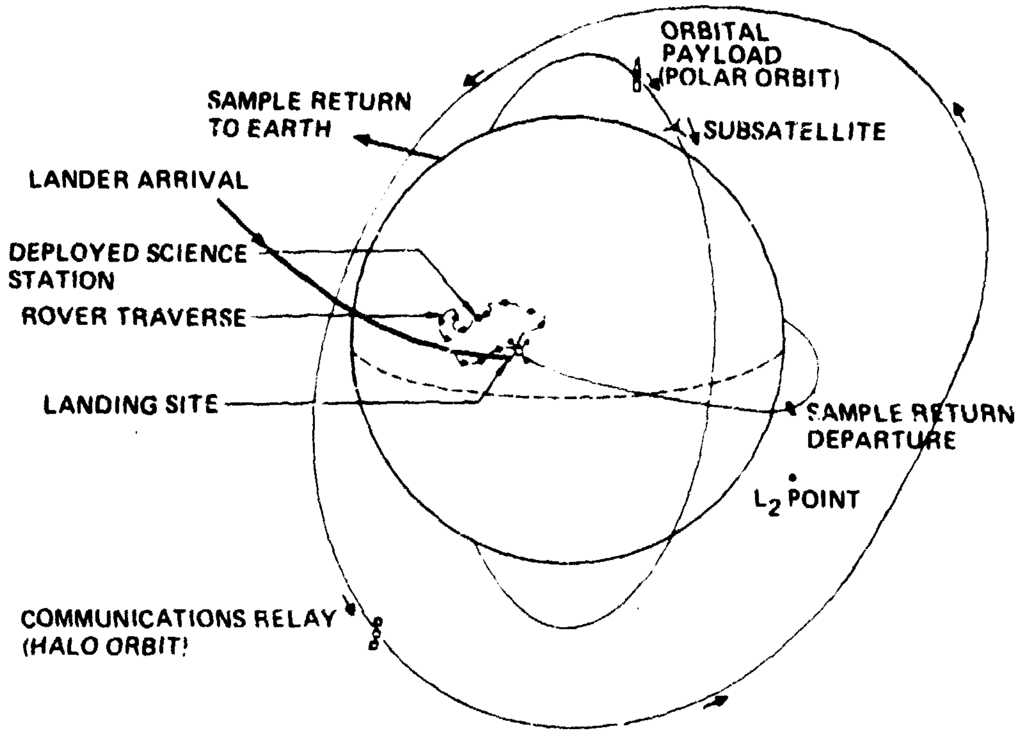

Figure 10. Automated Lunar Operations

performs various scientific functions including sample collection during the traverse. Samples are brought by the rover to the return system mounted on the lander platform; a departure to Earth by the sample return is shown.

Automated lunar payloads range from 500 to 900 kg (1100 to 2200 lbs) in lunar orbit and 1600 to 3700 kg (3500 to 8000 lbs) landed. These missions were found to be within the capability of the baseline STS (shuttle and upper stage).

2.8 AUTOMATED PLANETARY EXPLORATION

Three missions were used to characterize this program area. A wide variety of programs can be imagined; the three selected are intended only as representative of a class of requirement and do not constitute recommendations as to particular missions that should be carried out. The three example missions are:

- Automated Mars Surface Exploration and Sample Return
- Jupiter Atmosphere Buoyant Probe and Satellite Relay
- Automated Ganymede Lander

2.8.1 Mars Surface Sample Return

The mission goal of Mars Sample Return is exploration of the Martian surface with return of *10 kg (22 lbs) of surface and atmospheric samples.*

Figure 11 illustrates typical operations on the Martian surface. The lander stands on its legs with the two rover ramps deployed. One rover is on the ramp; the second is near the sample return system receptacle. This receptacle brings the sample canister to its launch location in the return system which is mounted on the lander. The lander serves as launch pad for the sample return ascent stage.

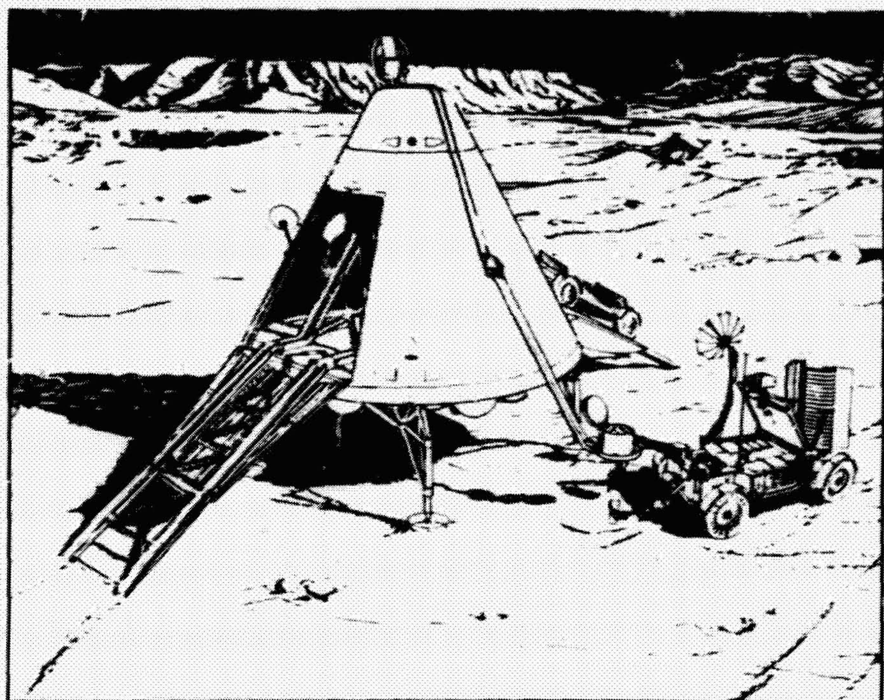

Figure 11. Mars Surface Sample Return

Table 4. MSR Transportation Requirement Summary

Option	Earth Return Mode	Mission	Lander Mass		Earth Departure ΔV		Mars Arrival ΔV	
			kg	lb	m/sec	ft/sec	m/sec	ft/sec
1	Atmos. entry	Fast	20 700	45,540	5 000	16,400	5 400	17,700
2	Atmos. entry	Slow	7 500	16,500	4 300	14,104	2 200	7,200
3	Earth parking orbit	Slow	18 400	62,480	4 300	14,104	2 200	7,200

Four mission mode options were investigated; the three described in Table 4 were retained.

2.8.2 Jupiter Buoyant Probe

Mission objectives for the Jupiter buoyant probe are the acquisition of data from the region of the upper Jovian cloud system over one day-night period (10 hours), with further one-time data acquisition to depths of at least 500 km (270 n.mi.) below the cloud tops.

Long duration mission data is collected by a science platform supported by a balloon. In Figure 12 it is shown just above the cloud tops. The deep probe has just been released and will descend into regions of increasing pressure and temperature. It relays data to the buoyant probe, for transmission to an orbital relay and thence to Earth.

It is possible that the Jovian atmosphere is too violent (shears and gusts) to allow operation of a balloon. Alternative mechanizations might be visualized. Clearly, additional data on the planet are needed to allow firm selection of a design approach.

The payload delivered to a synchronous elliptic Jupiter orbit (period 9.92 hours) is 2410 kg (5300 lb) of which 1305 kg (2875 lb) is the entry probe; the remainder is the orbital relay. This is a high delta v mission requiring a C_3 of at least 80 km^2/sec^2 (8.61×10^8 ft^2/sec^2) for Earth departure (Delta v for departure from a 500 km (270 n.mi.) orbit is about 6500 m/sec (21,325 ft/sec)), and 9100 m/sec (29,900 ft/sec) delta v for Jupiter capture.

Figure 12. Jupiter Buoyant Probe

2.8.3 Ganymede Lander

Mission goals are:

1) Orbital observation for Ganymede surface

2) Soft landing, with sample collection and analysis, stereo TV, plasma physics, magnetometer, seismic instruments.

The basic mission plan is to enter Jupiter space following a flight from Earth of approximately 800 days (C_3 of 80 km^2/sec^2 = 8.61×10^8 ft^2/sec^2). A Ganymede powered swing-by maneuver is used to reduce the ΔV required at Jupiter arrival. A series of Ganymede fly-bys follow (Ganymede pumps) reducing the orbital period. Four fly-bys take place before the landing; more could be used, but with diminishing return. The payload, including propellant for Jupiter space maneuvers, is 5910 kg (13,030 lb), injected to Jupiter transfer.

The lander system is based on the Surveyor vehicle, with the solar cell arrays replaced with RTG's. Also some changes in insulation are required, resulting in a modest increase in weight. The orbital element was based on the planned Mariner Jupiter Orbiter (MJO) 1981 spacecraft.

2.9 NUCLEAR WASTE DISPOSAL

Production of nuclear energy by fission results in production of highly radioactive wastes consisting of fission products and radioactive, mostly nonfissionable isotopes created by neutron capture. Nuclear waste material is initially millions of times more radioactive than the parent uranium. It decays, roughly exponentially, to a level below that of the parent in about 10 million years. There is no general agreement on the length of time required for decay to a "safe" level. Estimates of various authors range from "hundreds of years" to "millions of years."

A number of methods for permanent (i.e., not requiring a continuing monitoring program) disposal have been studied. One of these is disposal in space.

The power produced per unit of waste product produced by nuclear energy is large, leading to an economic leverage potentially large enough to permit the cost of space disposal. Most studies of space disposal, in order to enhance the economics, have dealt with refined wastes (short-lived isotopes separated out and held in monitored storage until they decay to safe levels). A system to dispose of total waste would be desirable if feasible.

Three options were initially considered. The second option, "partially refined waste" combined most of the disadvantages of options 1 and 3 and has been dropped. Option 1, refined waste as defined by a Lewis Research Center study, is the baseline for this study. Option 2 (formerly option 3) is total waste disposal.

The baseline option 1 waste package has a total mass of 3245 kg (7150 lb); about 50 such packages per year must be launched to support a representative nuclear power generation program. Two destinations were considered: 1) a 0.9 AV solar parking orbit, total delta V from low Earth orbit about 5115 m/sec (16,800 ft/sec), and 2) Solar system escape at C_3 = 150 km^2/sec^2 (1.61×10^9 ft^2/sec^2), requiring a delta V of 9160 m/sec (30,000 ft/sec).

2.10 SATELLITE ENERGY SYSTEMS

Recent studies have indicated the potential feasibility of collecting solar energy in space on a large scale, converting it to microwave power, and beaming the converted energy to Earth where it can be reconverted to electricity for commercial sale. Solar photovoltaic and heat engine (shown in Figure 13) converter satellites have been proposed.

Operational satellite design concepts range from 20×10^6 kg to 100×10^6 kg (44×10^6 to 220×10^6 lb) depending on power level, type of conversion, and technology level assumed. These satellites would be placed in a geosynchronous operational orbit by a process of piecemeal launch from Earth, assembly in a low orbit, and transfer to the operational orbit. Present indications are that Earth launch capability should be about 250 000 kg to 500 000 kg (550,000 lb to 10^6 lb) per launch, with a payload bay as large as is practicable.

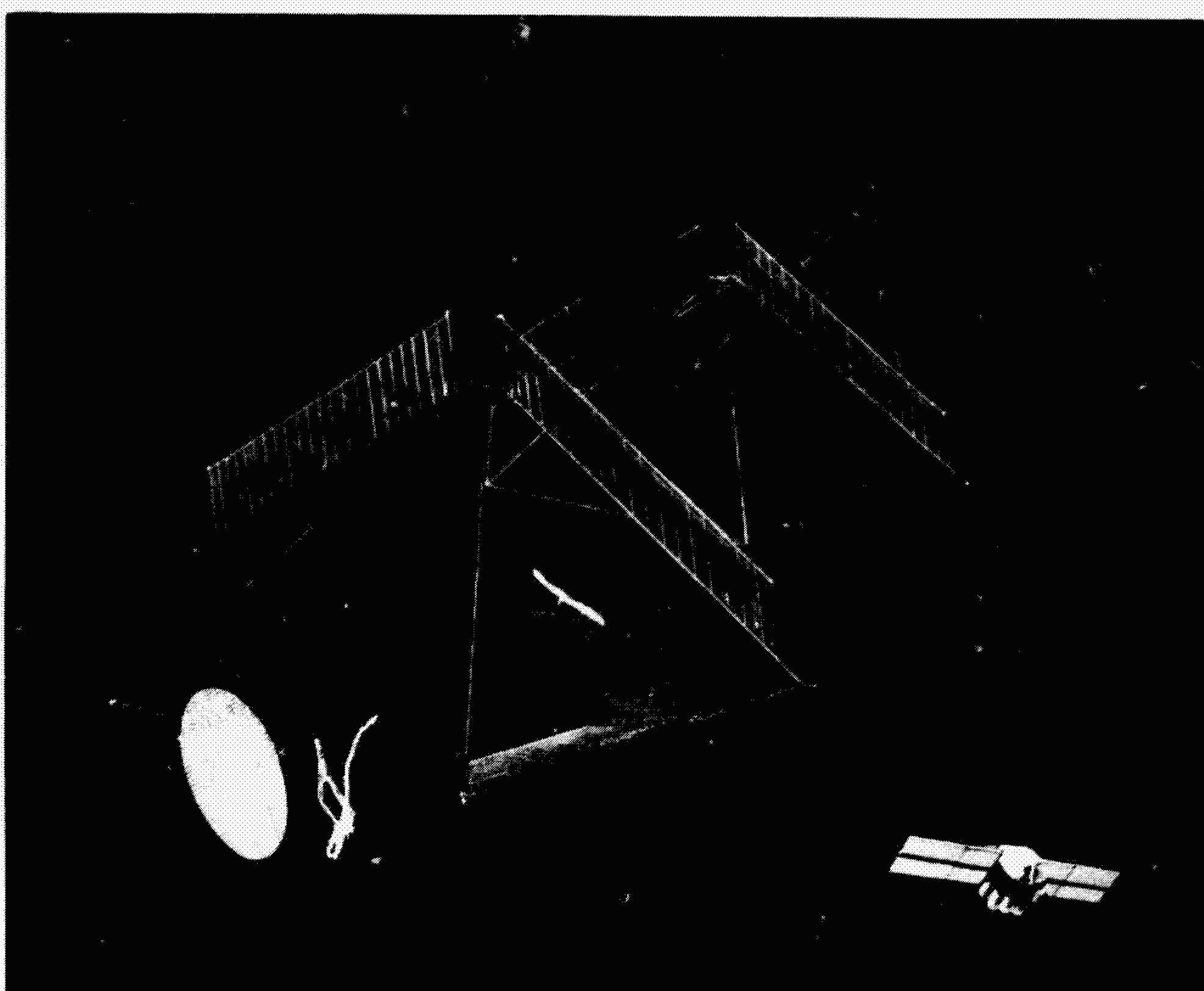

Figure 13. Satellite Energy System

The practicality of such a system is dependent on achievement of low costs for space transportation the HLLV and FSTSA studies have indicated promising approaches to achieving these low costs. Also critical is the operational capability to assemble large structures and systems in space, followed by system checkout, troubleshooting, and operation. Because a large extrapolation of current operational state-of-the-art is required to achieve capability to assemble and deploy power satellites, a demonstration laboratory and pilot plant program phase is indicated. This, in turn creates potential transportation requirements; a representative pilot plant is estimated as 250 000 kg (550,000 lb).

Various studies and projections of the rate at which power satellites should be brought on line have ranged from about 5000 megawatts to over 50,000 megawatts per year, corresponding to a range of one satellite (representative 10,000 megawatt size) every two years to five satellites per year.

2.11 SUMMARY OF MISSION-IMPOSED TRANSPORTATION REQUIREMENTS (Payload Only)

Transportation requirements in terms of total or annual payload mass delivered to the mission destination are compared on a logarithmic scale in Figure 14. Propellants or transportation vehicles required are not included. Annual requirements are shown where a crew rotation or resupply requirements exist. For nuclear waste disposal, the annual requirement reflects 50 waste packages per year; for satellite energy systems, placement of one power satellite per year in geosynchronous orbit is assumed. Table 5 compares payload requirements on a more detailed basis.

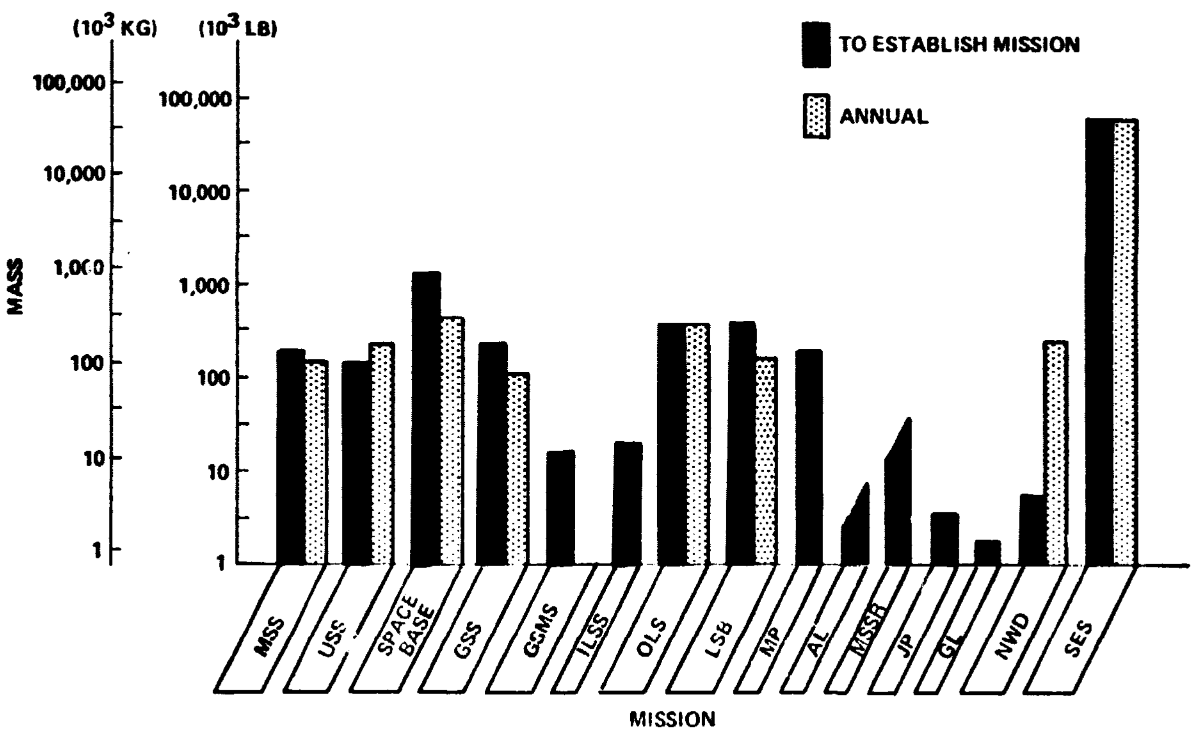

Figure 14. Mission Imposed Transportation Requirements Payload Only

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Table 5. Mission Requirements Summary

Program option	Mission option	Initial payload delivery requirements (e.g., station)				Crew rotation & resupply					Representative orbit transfer ΔV	
		Mass		Largest module		Typical size		Delivery mass	Return mass	Interval	Delivery	Return
		Per module	Total	Dia	Length	Dia	Length					
Low Earth orbit	12-man modular	12,400 (27,340)	98,800 (219,000)	4.3 (14)	11.8 (38.7)	4.3 (14)	13 (42.6)	15,000 (33,200)	11,000 (24,600)	3 months	N/A	N/A
	12-man unitary	50,000 ''10,000)	73,401) (184,000)	8.2 (27)	27 (90)	4.3 (14)	13 (42.6)	15,000 (33,200)	11,000 (24600)	3 months	N/A	N/A
	60-man base	100,000 (220,000)	580,000 1,280,000	8.2 (27)	32 (104)	4.3 (14)	13 (42.6)	20,000 (44,000)	11,400 (25,200)	3 weeks	N/A	N/A
Geosynchronous operations	Satellite maintenance sortie	6,570 (14,485)	6,570 (14,485)	4.3 (14)	4.8 (15.8)	N/A	N/A	N/A	N/A	N/A	4,815 (15,800)	4,405 (14,450
	8-man station (modular)	20,500 (45,400)	125,000 (275,000)	4.3 (14)	12.8 (42)	4.3 (14)	15.6 (51)	25,000 (55,000)	15,000 (33,000)	6 months	4,335 (14,220)	4,405 (14,450)
	50-man station	98,000 (215,000)		8.2 (27)	30 (98)	4.3 (14)	20 2 (66) modules	40,000 (88,000)	23,000 (51,000)	2 months	4,335 (14,220)	4,405 (14,450)
Manned lunar	Independent sortie	12,500 (27,500)	12,500 (27,500)	4.3 (14)	6 (20)	N/A	N/A	N/A	N/A	N/A	4,165 (13,665)	4,195 (13,765)
	Orbiting lunar station	15,000 (33,000)	210,200 (463,000)	4.3 (14)	12.8 (42)	4.3 (14)	10 (33)	60,000 (132,000)	6,120 (13,450)	109 days	4,205 (13,800)	4,195 (13,765)
	6-man, 6-month base	6,000 (13,200)	50,000 (110,000)	4.3 (14)	10 (33)	N/A	N/A	N/A	N/A	N/A	4,205 (13,800)	4,195 (13,765)
	12-man base	4,200 (9,250)	140,000 (310,000)	4.3 (14)	9.1 (30)	4.3 (14)	9.1 (30)	80,000 (176,000)	7,300 (16,000)	164 da's	4,205 (13,800)	4,195 (13,765)
	200-man base	20,000 (44,000)	3.5×10^6 (7.7×10^6)	6.6 (22)	20 (66)	6.6 (22)	20 (66)	135,000 (300,000)	12,000 (26,500)	1 month	4,205 (13,800)	4,195 (13,765)
Manned plar ntary	Manned Mars landing	50,000 (110,000)	110,000 (243,000)	6.6 (22)	15 (49)	N/A	N/A	N/A	N/A	N/A	6,871 (22,500)	5,552 (18,215)
Automated lunar	Polar orbiter	515 (1,135)	515 (1,135)	2.1 (7)	3 (10)	N/A	N/A	N/A	N/A	N/A	4,100 (13,950)	N/A
	Halo orbiter	900 (2,000)	900 (2,000)	3 (10)	4.6 (15)	N/A	N/A	N/A	N/A	N/A	3,490 (11,450)	N/A
	Sample return	1,635 (3,600)	1,635 (3,600)	4.4 (14.5)	3.4 (11)	N/A	N/A	N/A	N/A	N/A	6,220 (20,400)	2,900 (9,515)
Automated planetary	Mars sample return	7,500 (16,500)	7,500 (16,500)	4.4 (14.5)	4 (13)	N/A	N/A	N/A	N/A	N/A	6 500 (2),325)	6,400 (21,000)
	Jupiter buoyant probe	2,400 (5,300)	2,400 (5,300)	3 (10)	5 (16.4)	N/A	N/A	N/A	N/A	N/A	15,900 (52,200)	N/A
	Gauymede lander	5,900 (13,000)	5,900 (13,000)	3 (10)	6 (20)	N/A	N/A	N/A	N/A	N/A	6,500 (21,325)	N/A
Nuclear waste disposal	0.9 AU	3,245 (7,150)	3,245 (7,150)	2 (6.6)	1.8 (6)	N/A	N/A	N/A	N/A	N/A	5,115 (16,800)	N/A
	solar system escape					N/A	N/A	N/A	N/A	N/A	3,360 (30,700)	N/A
Satellite energy systems	Pilot plant/lab	25,000 (55,000)	250,000 (550,000)	TBD	TBD	Shuttle	Shuttle	TBD	TBD	TBD	N/A	N/A
	Orbital assembly station	TBD	TBD	TBD	TBD	TBD	TBD	TBD	TBD	TBD	TBD	TBD
	Operational satellite	TBD	$50\text{-}100\times10^6$ $(110\text{-}220\times10^6)$	TBD	TBD	TBD	TBD	TBD	TBD	TBD	6,000 (19,700)	(N/A)

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3.0 TRANSPORTATION SYSTEMS

3.1 MISSION/TRANSPORTATION OPERATIONS AND TYPES OF SYSTEMS

Operations were divided into three categories for analysis, as shown in Figure 15: 1) Earth launch and recovery, 2) orbit transfer, and 3) lunar or planetary transport (landing from the ascent to orbit). This is the most practical and efficient allocation of functions for vehicles employing the levels of technology assumed in this study. It is possible in principle to imagine a vehicle that would, for example, take off from Earth, travel to the moon, land, and return, but such a vehicle would require propulsion technology not definable today in engineering terms.

Transportation systems required to perform the investigated future missions include Earth launch vehicles, orbit transfer vehicles and landing/ascent vehicles.

Earth launch vehicle candidates include the space shuttle, a partly reusable heavy lift launch vehicle (HLLV) consisting of modified shuttle components such as the external tank and SRB, a fully reusable low cost heavy lift vehicle (LCHLV).

Orbit transfer vehicles and lunar or planetary landing/ascent vehicle candidates include fully reusable and partly reusable systems and both LO_2/LH_2 and LO_2/MMH propellants. LO_2/LH_2 is representative of deep cryogenic, high performance combinations. LO_2/MMH is representative of dense but lower-performance systems.

Also analyzed for some missions were more advanced orbit transfer vehicles, characterized by nuclear-Nerva (hydrogen-heater graphite reactor), nuclear-electric, and solar-electric systems.

3.2 TECHNOLOGY BASE

Technology base assumptions and guidelines were developed for each subsystem.

3.2.1 Main Propulsion

Chemical propulsion systems draw heavily on the technology produced by the shuttle SSME. High chamber pressures (20 MH/m^2; 3000 psia) for the larger engines are assumed in order to minimize engine envelopes; particularly important for small diameter multi-engine stages. Lower chamber pressures are assumed at lower thrust levels because of the problems in developing a small high pressure staged combustion engine. The lower chamber pressure assumed for the LO_2/MMH systems is

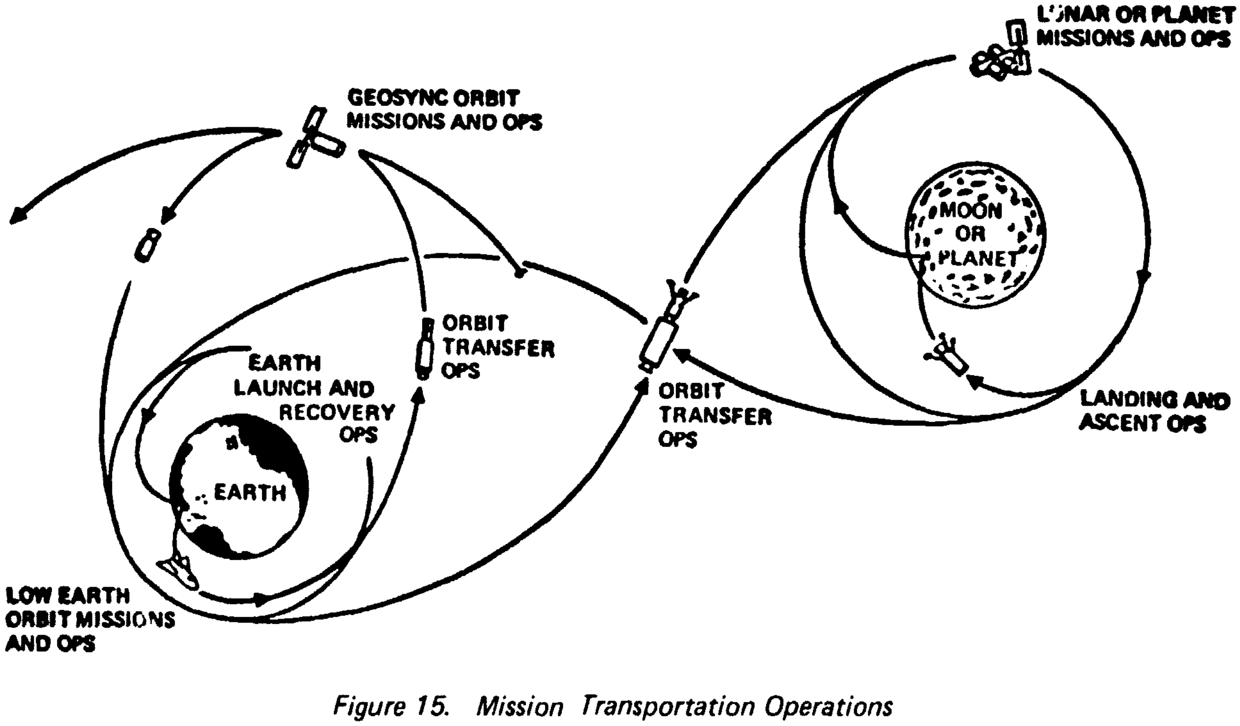


Figure 15. Mission Transportation Operations

related to the propellant being less effective in cooling. Vacuum Isp for LO_2/LH_2 engines was assumed as 462 sec, and for LO_2/MMH engines as 366 sec.

For nuclear propulsion, a Nerva type reactor appears most practical. An Isp of 780 sec. was used as an overall average considering start/stop losses and cooling in a reusable system. Nuclear-electric systems were assumed to use a heat pipe cooled reactor with a high temperature Brayton conversion system. Electric thrusters could be electrostatic or MPD types with argon as propellant.

3.2.2 Auxiliary Propulsion

The use of hydrazine monopropellant was baselined since inert mass reductions associated with more advanced systems such as storage bipropellants or O_2/H_2 are minimal when considering total system impact. More advanced systems would be more costly. An Isp of 220 seconds was used.

3.2.3 Structures

Graphite-plastic matrix composites were assumed for unpressurized main structures in reusable vehicles; aluminum skin/stringer was assumed for expendable vehicles or expendable tanks. Elevated temperature materials are assumed where normal working temperatures for aluminum or composites are exceeded. For example, structural elements of the nuclear electric tug would be titanium due to thermal radiation from hot parts and the heat rejection radiator.

Aluminum was assumed for all propellant tanks. Whenever possible, integral stiffening of the pressure vessel sidewall was assumed rather than suspended tanks. Reusable heat shields assumed shuttle technology where applicable; water cooled or other special heat shields were used where circumstances merit a departure from shuttle technology.

3.2.4 Thermal and Meteoroid Protection

Multilayer metallized plastic film (MLI) insulation was assumed for thermal protection of all main propellant tanks. A metal skin, non-structural for vehicles with integral tanks, was assumed external to the MLI, thick enough so that, in conjunction with the MLI, it provides sufficient meteoroid protection.

3.2.5 Electric Power

Fuel cells and batteries were assumed for electric power except for electric propulsion primary power. Fuel cells were tailored to the application and based on shuttle technology. Batteries were assumed to be Ni-Cad.

3.2.6 Avionics

LSI circuit chip technology was assumed available for data processing hardware; data bus techniques were assumed to minimize wire mass. Communications and GN&C systems assumed shuttle and full-capability tug technology levels. Laser radar was assumed available for rendezvous as required.

3.3 SUMMARY OF FINDINGS AND RESULTS TO DATE

The following findings summarize the key results of the study to the present:

- Missions fall into four transportation categories
- Reasonable transportation solutions exist for all missions
- Space-based power requires a tailored low recurring cost transportation solution

The categorizing of missions was an expected result, except for the apparent uniqueness of the power satellite mission. The identification of reasonable solutions is significant and is discussed in some depth on the pages to follow. Power satellite transportation tends to be unique. The requirements of an operational power satellite program are enough more demanding than the other missions to merit a tailored solution. Elements of this solution, if developed, could be used by other missions although the latter missions can be performed by systems requiring less development investment.

3.3.1 Mission Transportation Categories

The missions studied are listed in Table 6 according to their principal category. Some missions do not require transportation beyond the Space Transportation System presently under development. Most of the missions require some form of advanced orbit-to-orbit transportation. The nature of the requirement and size of vehicles varies considerably from mission to mission. Some missions require a Heavy Lift Launch Vehicle (HLLV) because of the mass or volume of payload elements; some of these

Table 6. Missions Fall Into Four Transportation Classes

• **Shuttle/IUS**	• **Current shuttle traffic model** • **Low earth orbit modular space station** • **Automated lunar exploration** • **Power satellite low orbit lab/demonstration phase**
• **Advanced OTV's** **(Note: Some OTV options require HLLV)**	• **Geosynchronous modular space station** • **Geosynchronous satellite maintenance sortie** • **Independent lunar surface sorties** } • **Orbiting lunar station** } **Also require LTV's** • **Lunar surface base** } • **Automated planetary exploration (advanced missions)** • **Nuclear waste disposal** • **Power satellite geosynchronous orbit demonstration phase**
• **HLLV**	• **Low earth orbit unitary station and space base** • **Geosynchronous unitary space station** • **Manned planetary exploration (also required Mars landing/ascent vehicle)**
• **Very low cost transportation**	• **Power satellite operational phase**

(Note: HLLV provides cost and operational benefits to some mission options that do not require it)

also require orbit-to-orbit transportation. The use of a HLLV benefits many missions that could be performed without it, as will be shown. Also it is important to note that some of the orbit transfer vehicles require a HLLV to launch them to low Earth orbit.

The special category of the power satellite operational mission results from the great mass delivery requirement and resulting need for lowest attainable costs. Demonstration laboratory pilot plant activities leading up to development of an operational system do not require the very low cost systems and would probably be carried out prior to development of such systems.

An additional dimension to the categorization and comparison of missions is the magnitude of the requirement. The missions are compared in Figure 16 in terms of the total annual launch requirement to low orbit, assuming LO_2/LH_2 orbit transfer systems except in the case of the power satellite, where self-power is assumed. Note the logarithmic scale. In several cases it was necessary to assume a number of missions per year in order to derive an annual requirement.

3.3.2 Characteristics and Evaluation of Transportation Solutions

Reasonable solutions were found to all mission requirements within the scope of technology presently understood. All of the transportation options employed technology levels such that advanced development could begin now if the need were clear.

Costs were found to be affordable in the context of carrying out the mission programs setting the requirement. Over the past few years, stimulated by the Shuttle developments, it has become clear

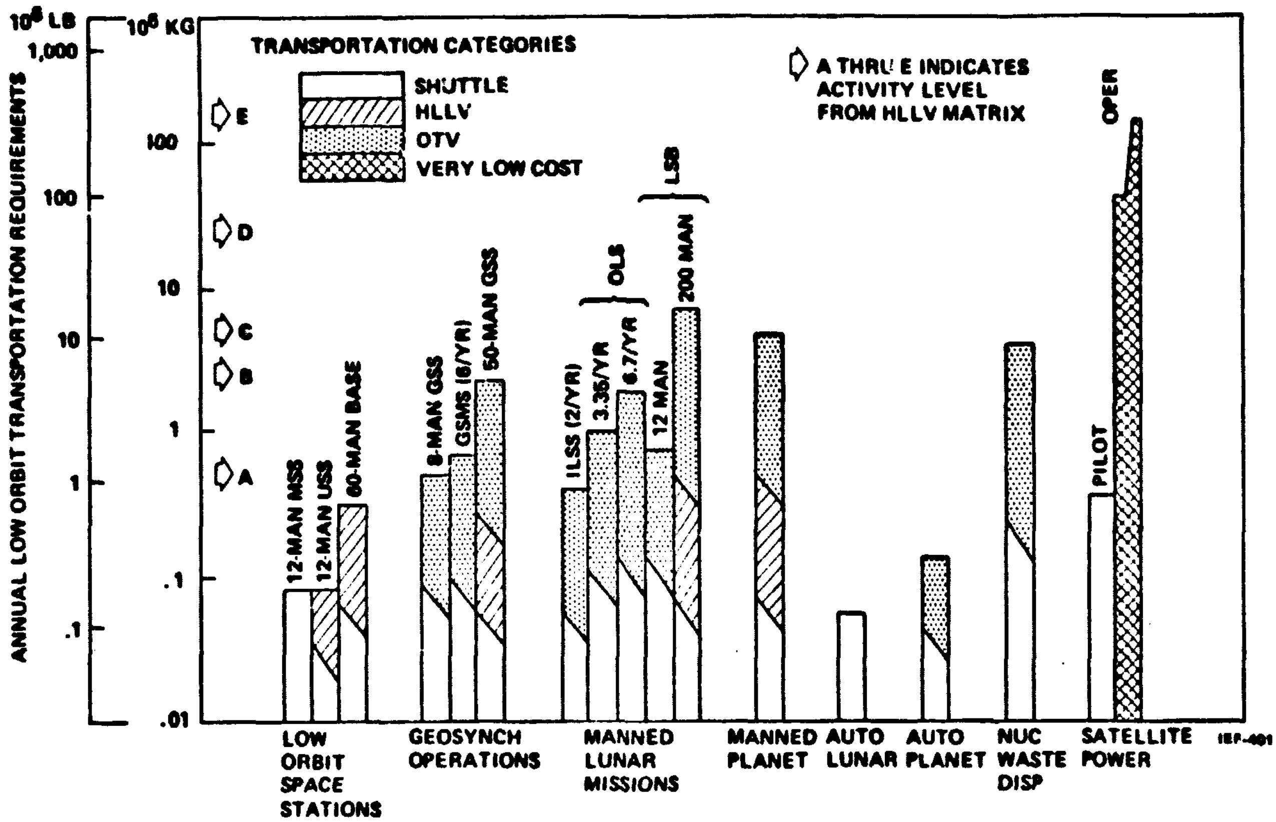


Figure 16. Transportation Task for FSTSA Missions

that major improvements are possible in space transportation technology, leading to cost effective systems through reusability and that this line of development can have as significant an impact on space transportation as the development of many of the ultra-high performance technologies (e.g., laser propulsion) frequently discussed.

The intrinsic cost of space transportation to low earth orbit is not high. The cost of the energy difference of a unit mass in a low orbit, as compared to at rest on the Earth's surface, is about 40¢/kg (18¢/lb) at typical current retail electrical energy costs. Since rockets are inefficient in converting propellant chemical energy to payload kinetic energy, the cost of propellant for an actual large rocket vehicle is about \$10/kg (\$4.50/lb) per unit payload mass. Even these values are very low compared to the roughly \$2000/kg (\$900/lb) achieved by expendable systems.

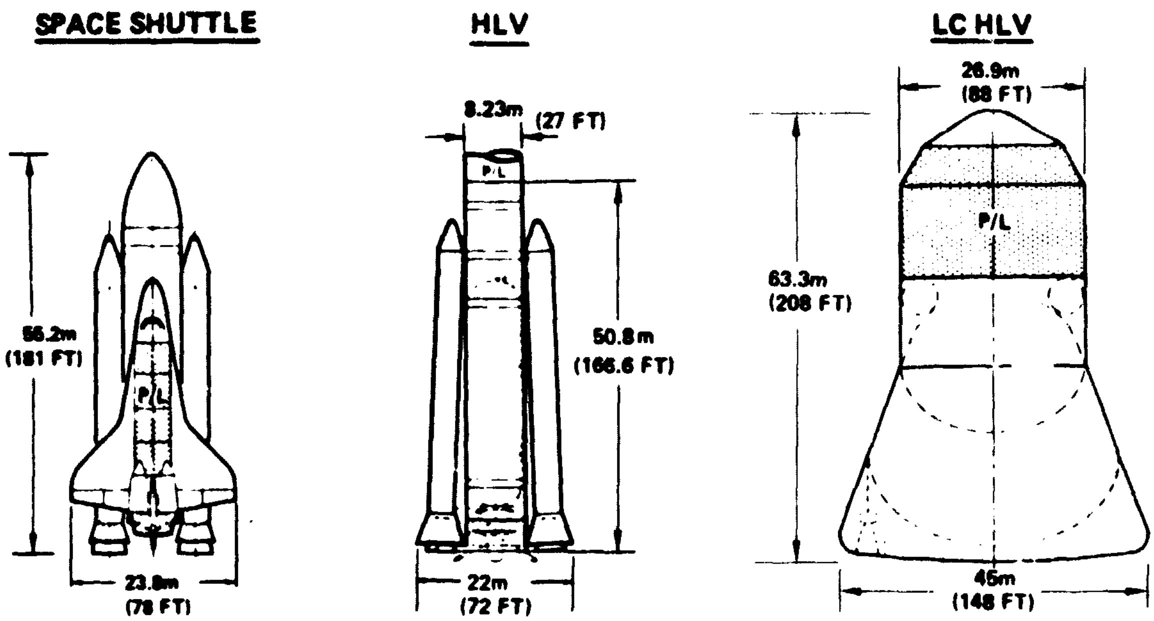
It is clear that the shuttle is the first step in a new generation of reusable technology. Indications are that reusability combined with high traffic rates can approach cost figures no more than two to five times propellant costs, a typical target being \$45/kg (\$20/lb).

3.3.2.1 Earth Launch Vehicles

The vehicles shown in Figure 17 are representative of Earth launch vehicle needs identified by the study. At the left is the space shuttle presently under development. Second is a shuttle-derivative HLLV employing shuttle engines, a modified external tank, and 2 or 4 shuttle SRB's. This class of HLLV is appropriate to all HLLV needs except the operational power satellite; the HLLV study has identified several potential configurations to satisfy the need.

The operational power satellite mission needs a low-cost space freighter with a payload of 200 000 kg (440,000 lb) or more. Shown is a single-stage ballistic fully reusable configuration. Two-stage ballistic/ballistic vehicles are also promising.

The cost of low orbit transportation was shown by the first quarter of the HLLV study to depend on



	Space Shuttle		HLV		LC HLV
• PAYLOAD @ 270 NM 28.5 DEG	25 000 KG* (55,000 LB)*	W/2 SRB	72 270 KG (159,000 LB)		227 000 KG (500,000 LB)
		W/4 SRB	110 910 KG (244,000 LB)		
• GLOW (10^6)	1.91 KG (4.2 LB)	W/4 SRB	3.2 KG (7.09 LB)		10.33 KG (22.7 LB)

* WITH ONE OMS KIT

Figure 17. Earth Launch Vehicle Candidates

configuration, propellant choice payload class, and most strongly on activity level. Costs for a few of the better performing configurations are compared in Figure 18. An idealized lower bound of cost versus activity level is also shown, indicating expected cost capability of a parametric HLLV tailored to each activity level. (In principle, an actual fixed vehicle could attain the lower bound at only one activity level; at any other level a different vehicle would do at least slightly better). In subsequent figures this idealized HLLV is termed a "rubber" HLLV; its cost characteristic curve is used in a parametric manner.

Using the rubber HLLV cost versus activity level from the previous figure, a parametric curve was developed showing annual cost for low Earth orbit transportation versus activity level (annual transportation requirement) as shown in Figure 19. The cost figures include a 20% increase factor for inefficiencies resulting from such things as packaging and propellant transfer losses. At the lower activity levels there is no cost advantage to a HLLV. FSTSA missions have been spotted on the curve; the automated lunar and low Earth orbit space stations are off the curve to the left. All of the missions are in HLLV activity level ranges A through C except the operational satellite energy systems (power satellites) at level E. These costs do not include direct cost of orbit-to-orbit systems but do include low Earth orbit transportation costs for their propellant. All of the transportation costs are affordable in the context of the kind of program environment that would lead to these missions.

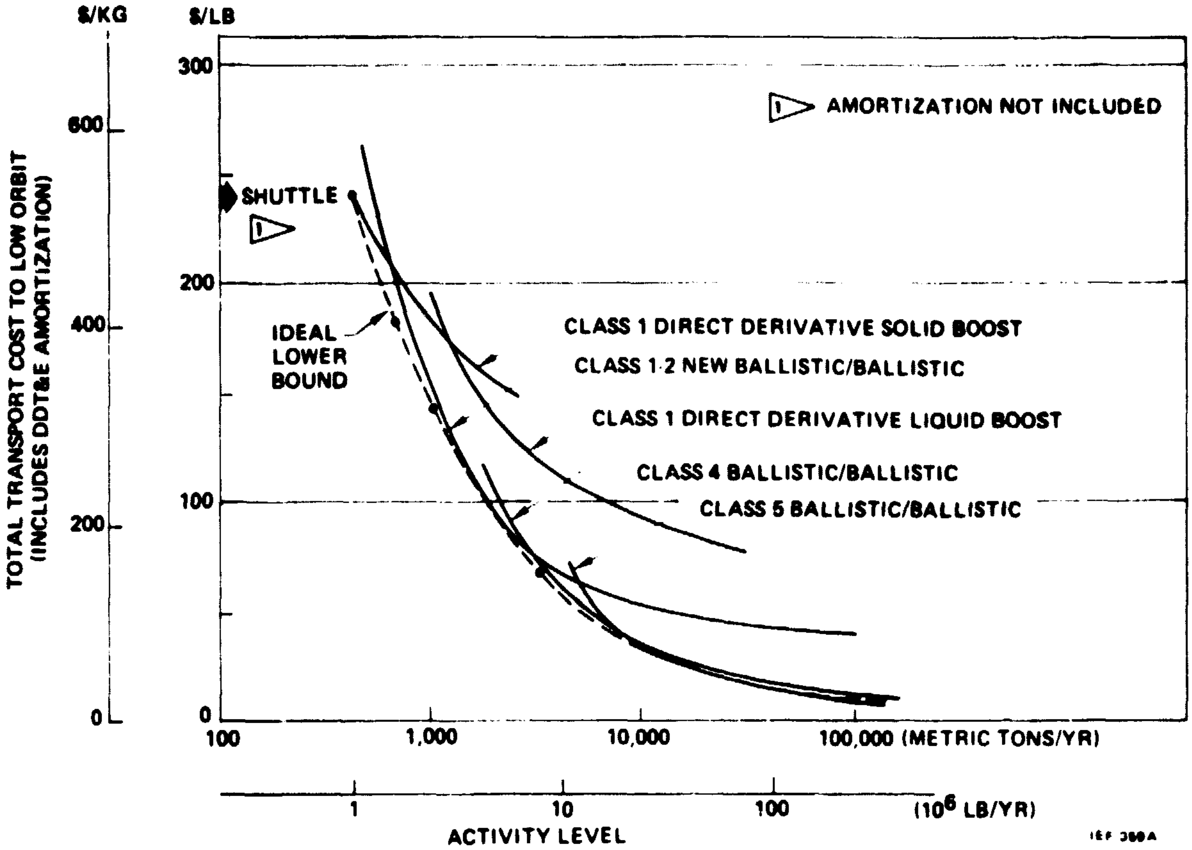


Figure 18. LEO Lift Costs From HLLV Study (First Quarterly Review)

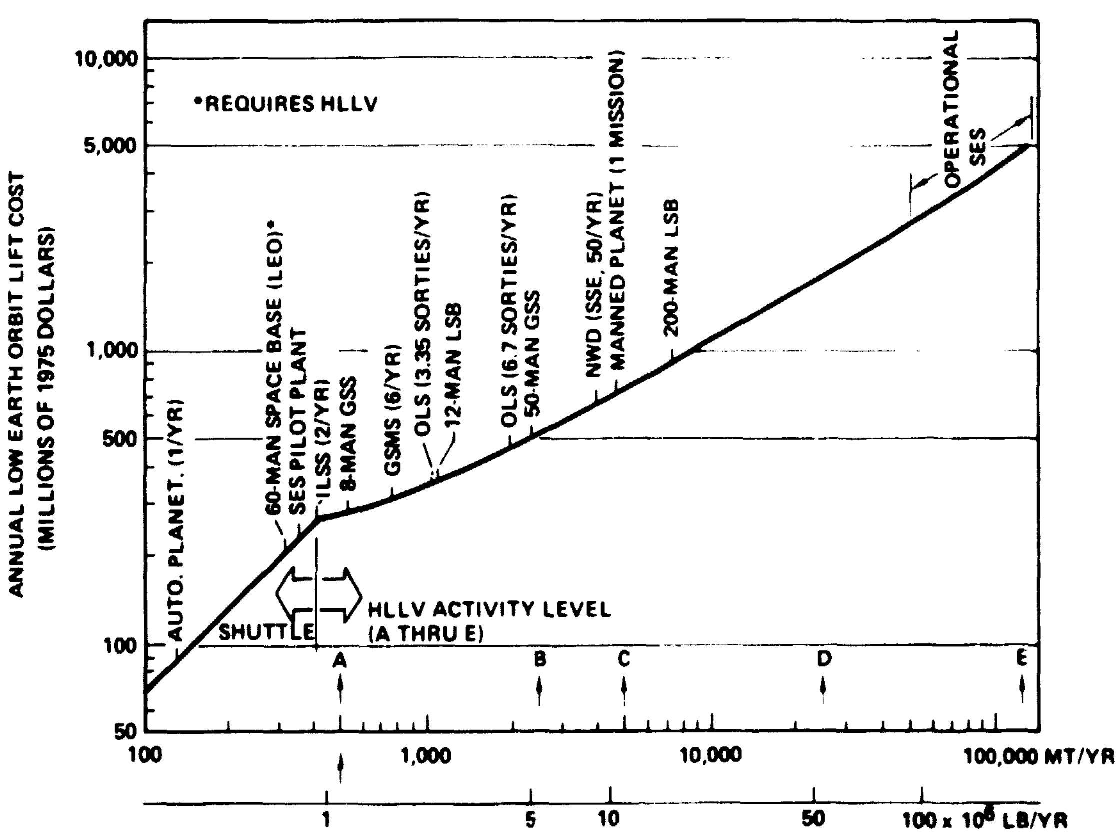


Figure 19. Lift Cost to Low Earth Orbit

3.3.2.2 Orbit Transfer and Lunar Transport Systems

The match-up-of transportation system options and missions studied during the Phase I Extension is illustrated in Figure 20. Transportation options are identified by the type of application, propellant option and staging method employed. Launch vehicles were not analyzed during this phase since a separate heavy lift launch vehicle study currently in progress provided sufficient data for the Extension phase.

SPACE TRANSPORTATION SYSTEMS

MISSIONS	OTV LO_2/LH_2 1 STG	OTV LO_2/LH_2 1½ STG	OTV LO_2/LH_2 COM STG	OTV LO_2/MMH 1 STG	OTV LO_2/MMH COM STG	OTV LO_2/MMH 2 STG	OTV NUC/LH_2 1 STG	OTV ELEC NUC 1 STG	OTV ELEC SOLAR 1 STG	LTV LO_2/LH_2 1 STG SM DIA	LTV LO_2/LH_2 1 STG LG DIA	LTV LO_2/LH_2 1½ STG	LTV LO_2/MMH 1 STG	LTV LO_2/MMH 1½ STG
GSS	X	X	X		X		X	X	X					
GSMS	X [2]		X		X									
ILSS			[3]		X	[4]				X		X	X	X
OLS	X	X	X		X		X			X	X	X	X	X
LSB	X	X	X		X		X			X	X	X	X	X
[5] M.P.	X						X							
[6] MSSR	X													
JBP	X	X												
GL	X			X										
NWD	X		X											
SES								X	X					

(APPLICATION: OTV, LTV; PROPELLANT OPTION; STAGING OPTION [1])

[1] ALL MANNED MISSIONS RETURN TO EARTH ORBIT UNLESS NOTED
[2] ALSO WITH AEROBRAKING AND DIRECT ENTRY
[3] COMMON PLUS DROP TANKS
[4] WITH DIRECT ENTRY
[5] CLUSTERING REQUIRED AND 3 STAGES
[6] DIRECT EARTH LANDING – SLOW RETURN

Figure 20. Mission/Transportation Options

The transportation options indicated, and their match-ups with the missions, primarily reflect the results of Phase I. In summary, there are 32 cases involving OTV's and 14 cases employing LTV's. Transportation options not included during this phase include planetary landers and two stage OTV's and LTV's. Planetary landers were excluded from further analyses since baseline systems were defined in Phase I and incorporated as part of the payload requirements for those missions. Two-stage OTV's and LTV's were also excluded from further analyses since Phase I results indicated no better performance than the 1 1/2 or common stage concepts for vehicles that expended a fairly costly portion of the total system. (The common stage OTV is a special case of a two-stage vehicle; the stages are equal in size and both are reused.)

The low Earth orbit space stations are not reflected in Figure 20 since space transportation beyond low Earth orbit is not required. Automated lunar missions are not shown because the Shuttle or Shuttle with IUS can provide the required payload capability.

All of the OTV concepts considered were analyzed for the geosynchronous space station (GSS) mission. Results are representative of those obtained for other missions. To illustrate the influence of different propellant and staging concepts on overall size and general arrangement, the OTV concepts are shown in Figure 21 all sized to perform the

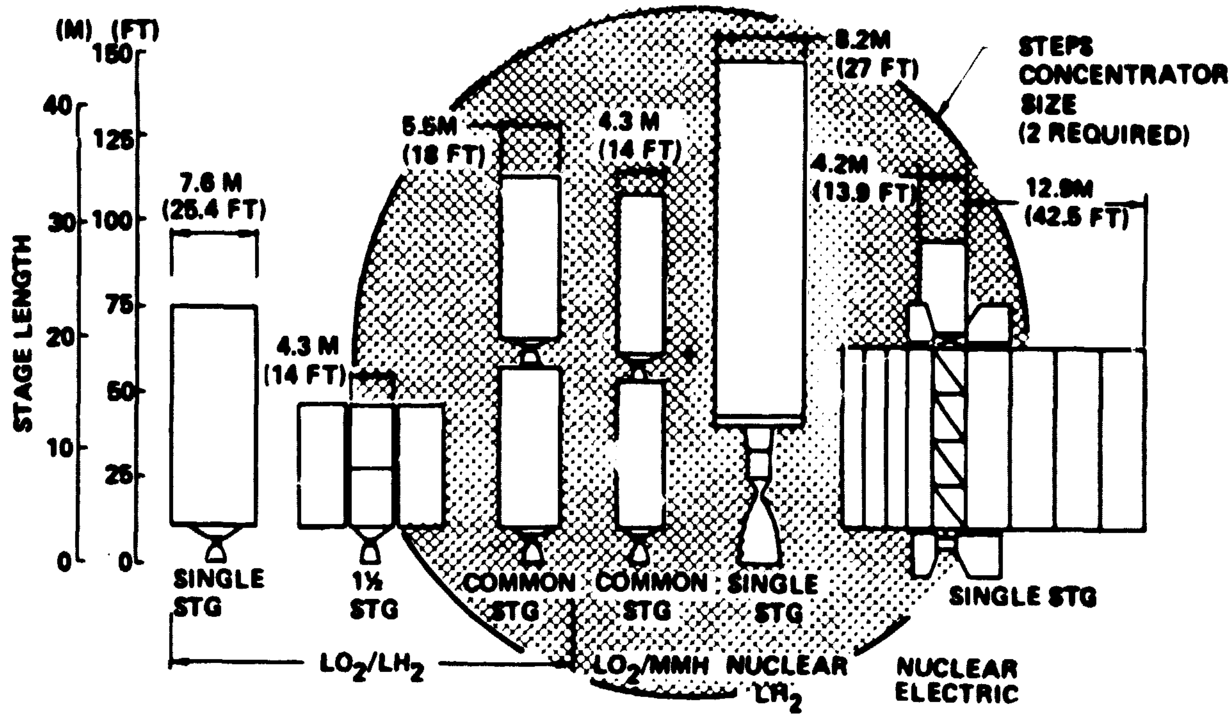


Figure 21. OTV Concepts and Size Comparison, GSS Mission

GSS crew rotation/resupply mission. Clearly these differences in configuration have a major impact on launch vehicle selection and orbital assembly operations.

The concepts illustrated are representative of the configuration arrangements regardless of the mission. In general, these concepts are being considered for the following reasons: the LO_2/LH_2 single stage for its simplicity; the LO_2/LH_2 1½ stage for its performance and being size compatible with the shuttle; the LO_2/LH_2 common stage for performance as well as complete reusability; LO_2/MMH stages offer size compatibility with the shuttle; nuclear LH_2 stages provide good performance and the nuclear electric and solar electric stages provide relatively low propellant refueling requirements. These last two concepts however, are only used for delivery of unmanned payloads due to their slow trip times; consequently, they are supplemented by small chemical stages for manned payloads.

The Solar Thermal Electric Propulsion System (STEPS) OTV identified by one of its large concentrators is further illustrated in Figure 22. It consists of two large solar collectors assembled in low orbit from shuttle compatible elements. Sunlight is concentrated into a cavity absorber to heat helium gas driving a closed Brayton turbogenerator producing electric power. Cycle waste heat is rejected by thermal radiators. Electric power is routed to the propulsion system employing MPD thrusters and argon propellant.

Vehicle Analyses—All configurations were analyzed in sufficient depth to develop the performance and cost data needed for comparative evaluation. The two (common) stage LO_2/LH_2 system shown in Figure 23 is used to illustrate the level of detail used in defining configurations and in subsequent charts to show representative weight and cost summaries.

The common-stage OTV concept consists of two nearly identical stages used in series to provide the required mission delta V. The first of these stages is used to provide approximately 85% of the delta V required for leaving Earth orbit on a crew rotation/resupply flight. Stage 2 provides the remainder of the boost delta V as well as that required for injection into the destination orbit and return. Following separation from stage 2, stage 1 is flown back into Earth orbit. Splitting the delta V in the above manner results in the stages having identical propellant capacities. Subsystem design approaches are also common between the stages including the

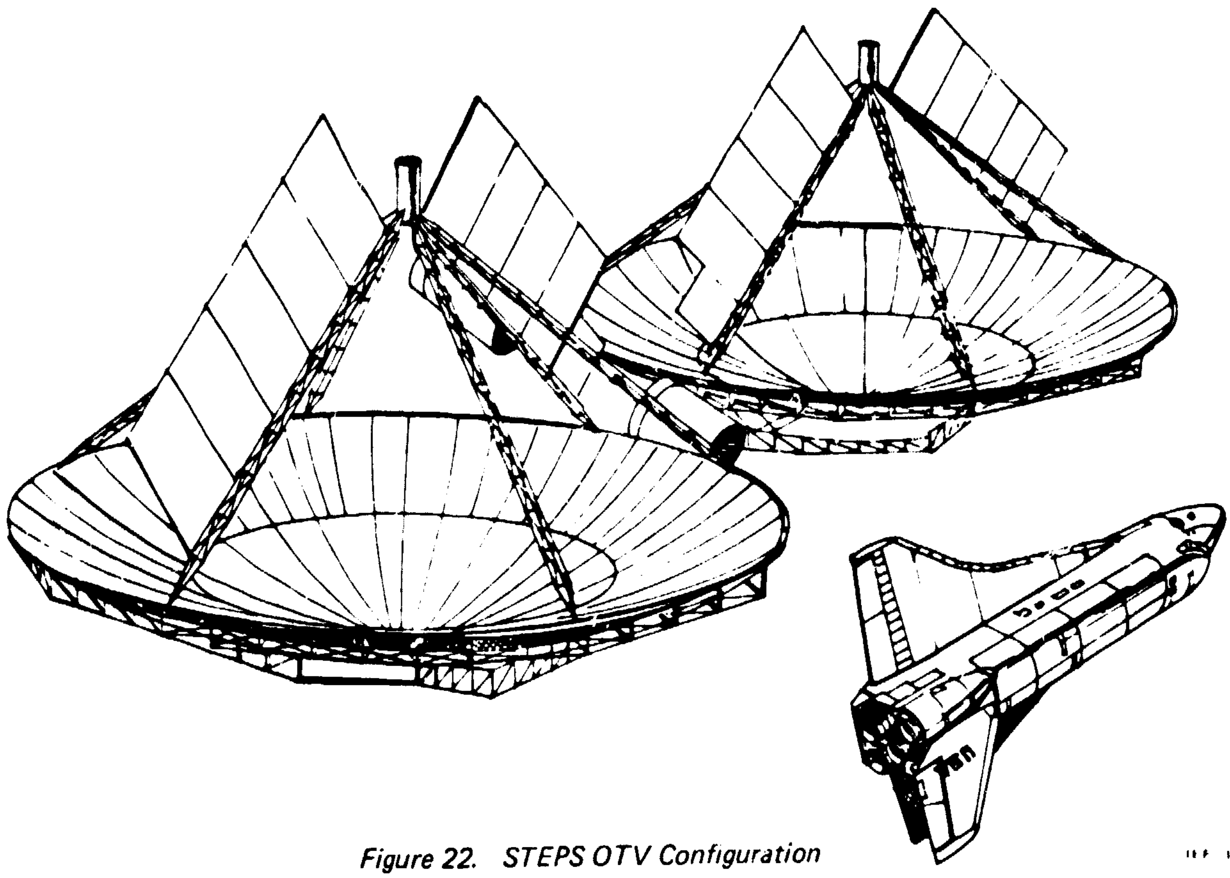


Figure 22. STEPS OTV Configuration

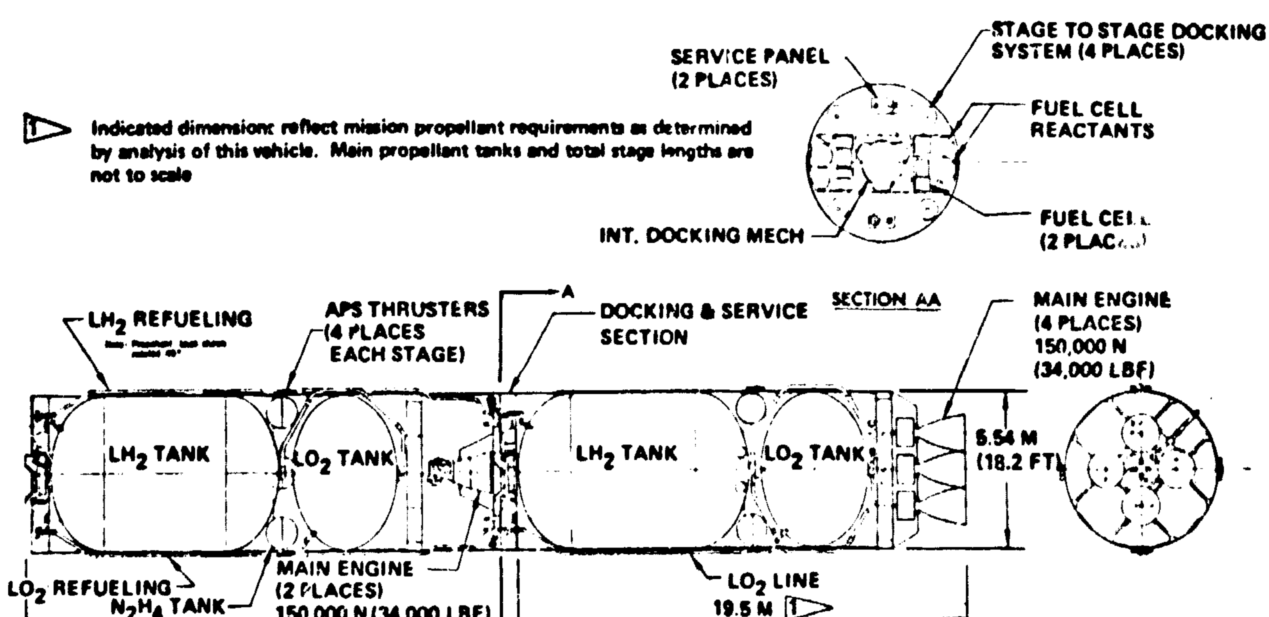


Figure 23. Common Stage LO_2/LH_2 OTV Configuration, GSS Mission

size of the main engine. Taken individually, each of these engines is similar to the single stage concept in terms of subsystem selection and location. When sized for the GSS mission, stage 1 is approximately 19.5 m (57.3 ft) in length. At the forward end of the stage are two types of docking provisions. One of these systems is used to connect with stage 2 while the center mounted unit is an international type design that allows docking with systems other than stage 2; examples include a tanker for independent servicing or a space station if basing is required while awaiting the return of stage 2. Four 150 000 n (34,000 lb) thrust engines are used on stage 1 to provide the desired T/W for Earth orbit departure. Sizing of this engine was the result of thrust requirements and engine out capability for stage 2.

Stage 2 is also 19.5 m (57.3 ft) in length; stage diameters are 5.5 m (18.2 ft). Docking provisions are required at both the forward and aft ends. The forward docking station uses an international unit for attaching payloads. In addition, this unit accommodates tankers or is used to connect the stage to a space station for basing. The aft docking provisions are used in conjunction with those in the forward section of stage 1 and enable the stages to be connected. Provisions are also included on stage 2 to allow servicing of stage 1 when the two stages are connected and the tanker is docked at the forward end of stage 2. Only two of the 150 000 n (34,000 lb_f) thrust engines are required on this stage.

The total startburn mass of the common stage LO_2/LH_2 OTV for the GSS mission is approximately 195 260 kg (430,480 lb).

Dry mass of the two stages is nearly identical as the mass of the two additional engines on stage 1 is balanced by stage 2 having heavier tankage for APS and electrical power reactants and additional structural mass due to the interstage surrounding the engines and supporting the stage to stage docking and service provisions.

The masses of the stages are based on the following factors: The body shell includes the forward and aft skirts as well as intertank structure and uses composite materials. The remainder of this structure is aluminum. The main engines use 20.6 MN/m^2 (3000 psia) chamber pressures while the pressurization system for stage 2 was sized for six main burns and four burns for stage 1. Cryogenic stored helium is used for preburn and GO_2 and GH_2 tapped off the engines during the burn. Hydrazine monopropellant is used for APS. The selected avionics system provides autonomous capability including rendezvous and docking. LO_2/LH_2 fuel cells provide an average load of 2 kw for the seven day mission of stage 2 and one day mission for stage 1. Rechargeable nickel-cadmium batteries are used for peaks with each sized for 50 amp-hrs. Approximately 0.25 cm (0.5 in) of multilayer insulation is used around the main tanks in addition to a purge system using helium. A mass contingency equal to 15% of hardware mass has been added to account for historical growth.

Main propellant loadings include 2% flight performance reserves; a 10% reserve is included in fuel cell reactants and APS propellant.

A mass comparison of the candidate transportation systems is presented in Figure 24 for the crew rotation/resupply flight (it requires a slightly larger stage than the delivery of one of the two station clusters). Both startburn and resupply masses are presented. The lower mass of the LO_2/LH_2 stage and common stage systems compared to the single stage approach is due to transporting less inerts as a result of staging. The LO_2/MMH common stage is heavier than those of the LO_2/LH_2 systems as a result of having a lower Isp (approximately 90 sec).

The nuclear LH_2 and two electric systems offer both the lowest startburn and resupply mass primarily as a result of their high Isp (nuclear LH_2 - 780 sec (avg) and nuclear and solar electric - 2500 sec). Included in the resupply mass of the electric systems is the propellant for the supplemental chemical OTV.

Cost Analyses—DDTE and first unit cost were developed at the cost element level indicated in Table 7 for all transportation concepts. These costs were derived using cost estimating relationships (CER), the mass summaries and costing groundrules peculiar to each option. All costs are expressed in 1975 dollars with no fee included. Program management includes only the contractors effort.

In the case of the common stage LO_2/LH_2 system, a total DDTE cost of approximately $420 million is predicted while the unit cost for a complete vehicle is approximately $54 million. The relatively

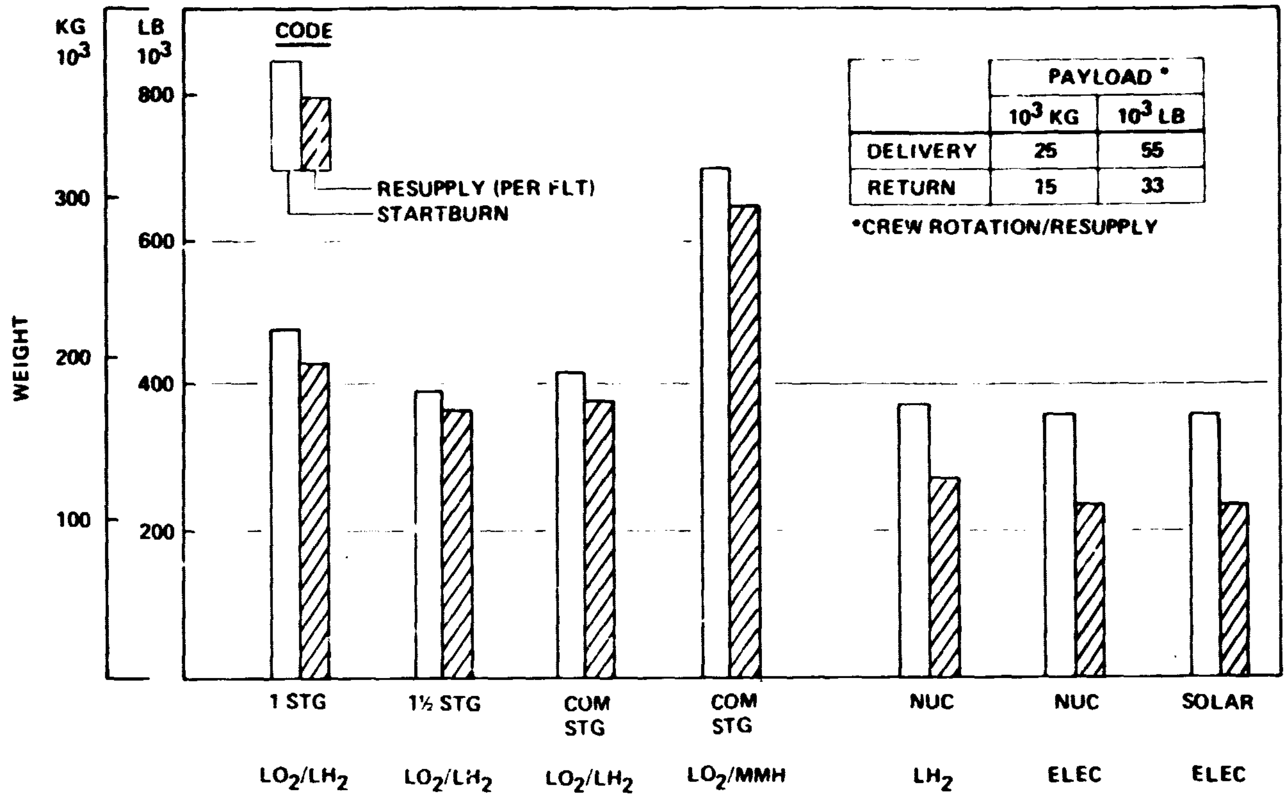


Figure 24. OTV Weight Comparison for GSS Mission

Table 7. Common Stage LO_2/LH_2 OTV Cost Summary, GSS Mission

Cost element	Stage 1 $ in millions DDT&E	Stage 1 $ in millions 1st unit	Stage 2 $ in millions DDT&E	Stage 2 $ in millions 1st unit
Flight hardware	(157)	(26.2)	(9)	(19.8)
Structures and mechanisms	8	2.6	1	1.7
Main propulsion	126	6.5	2	3.3
Auxiliary propulsion	5	1.8	2	1.8
Avionics	11	10.0	3	8.0
Electrical power	3	2.0	.9	2.6
Thermal control	4	2.5	.5	1.8
Assembly and checkout		.8		.6
Systems engineering and integration	(5)		(1)	
Software engineering	(3)		(1)	
Systems testing	(112)		(57)	
Ground test hardware	65		30	
Flight test hardware	39		25	
Test labor	8		2	
Ground support equipment	(8)		(2)	
Initial tooling	(.9)		(.8)	
Program management	(17)	(1.6)	(4)	(1.2)
Subtotal	302.9	27.8	74.8	21
Cost equivalent of mass contingency	33.3	2.8	8.2	2.1
Total	336.2	30.6	83.0	23.1

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low DDTE flight hardware cost associated with stage 2 is the result of its high degree of commonality with stage 1.

Development and unit cost of the options are compared in Figure 25. The high DDTE cost for the nuclear LH_2 and electric systems is the result of the expensive nuclear reactors. It should also be recognized that the cost of the two electric systems is based on data not nearly as well defined as that for the chemical systems. In addition, the electric systems will require orbital assembly and this cost has not been included. First unit cost of the nuclear LH_2 and nuclear electric systems is again high due to the reactors. The high unit cost of the solar electric system results largely from structural complexity.

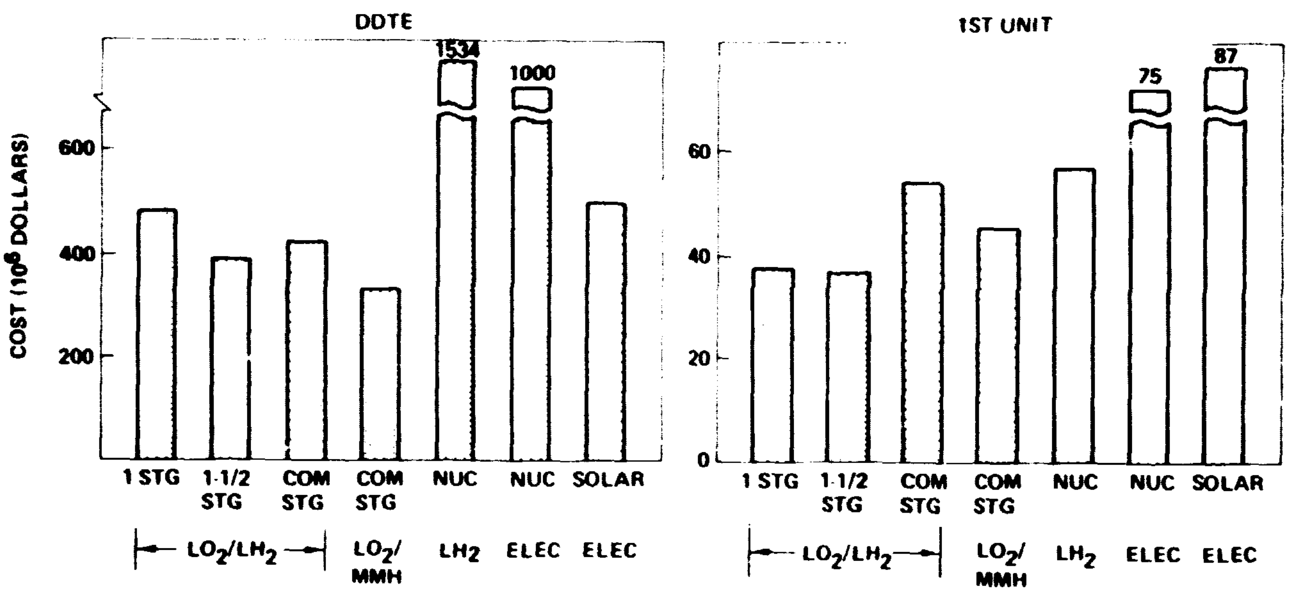


Figure 25. OTV Cost Comparison, GSS Mission

Evaluation—A total cost indicator for the OTV options as a function of the number of OTV flights was developed by using the DDTE and unit cost from the preceding page and a $100 per pound launch cost to cover delivery of the OTV hardware and propellant. As indicated by Figure 26, the common stage LO_2/LH_2 system becomes the cheapest after nine flights and provides a savings of approximately $200 million over the 1 1/2 stage system when a total of 60 flights are performed.

A further comparison of total use costs is shown in Figure 27. The composition of the OTV cost indicator is appropriate to 100 total uses, about equivalent to six uses per year over a 15 year period. This corresponds roughly to HLLV activity level B and to the 50 man geosynchronous station. It is cautioned that indications from this chart will not necessarily hold true for other activity levels or where a specific Earth launch vehicle is chosen. Several specific low orbit transportation cost slopes are shown, with the slope drawn such that sum of low orbit lift cost and OTV writeoff cost is constant. Also shown are curves of constant total cost using the rubber (continuously variable cost optimized) HLLV characteristics described earlier. Cost and mass values for the solar and nuclear electric systems include a small LO_2/LH_2 high thrust OTV, operating in conjunction with the electric OTV's, performing the crew transportation role. The electric OTV's deliver cargo and refueling propellant for the crew transport OTV's. The apparent cost advantage of the LO_2/LH_2 systems is evident. Absolute differences, however, are not large enough to justify discarding any of the options on the basis of this analysis.

Other factors can also be used to evaluate the various OTV concepts.

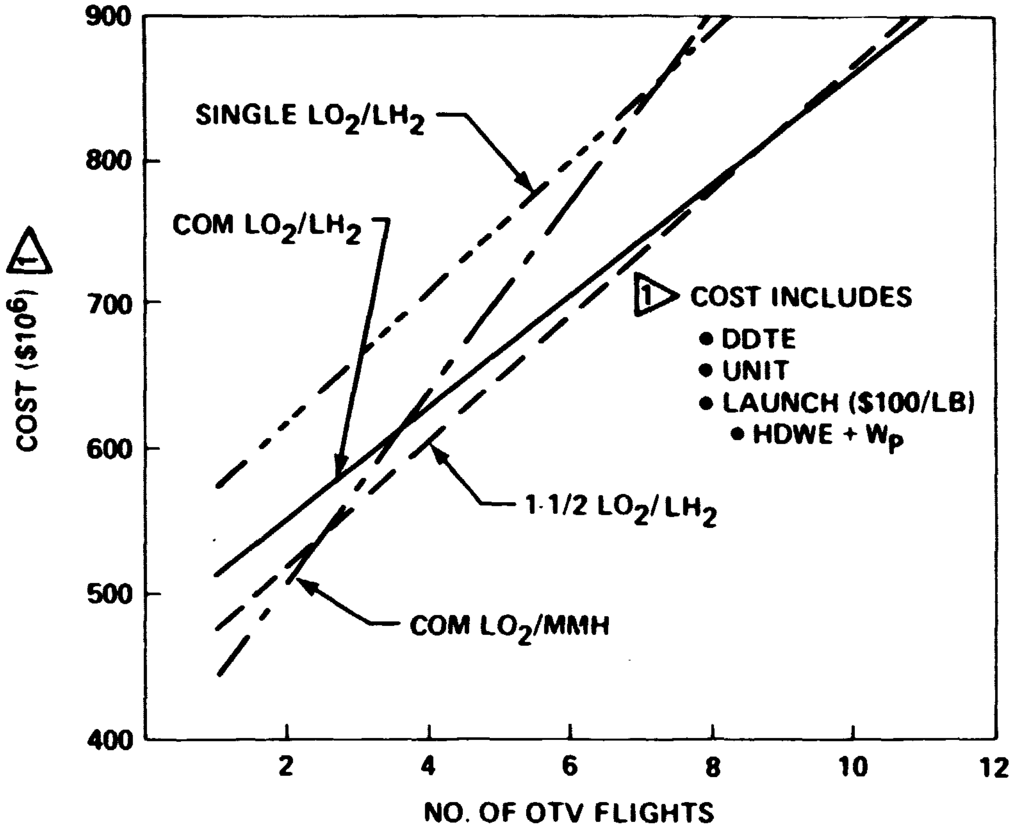
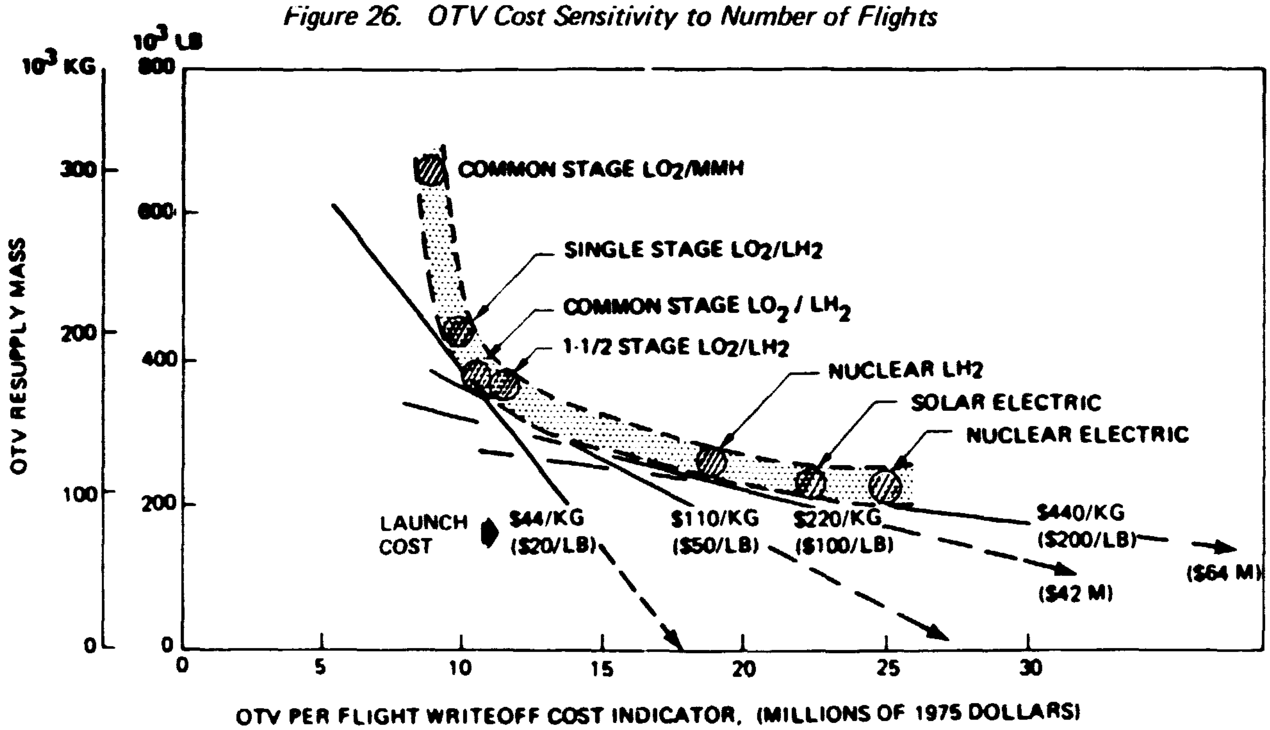


Figure 26. OTV Cost Sensitivity to Number of Flights



DEFINED AS 1% DDT&E COST, PLUS
10% (REUSABLE UNIT COST + INITIAL LAUNCH) PLUS
EXPENDABLE UNITS AT 50TH UNIT, 90% LEARNING

Figure 27. OTV Flight Cost Relationship to Launch Cost

There are no major concerns with the LO_2/LH_2 single stage. The LO_2/LH_2 1 1/2 stage requires either two or four drop tanks and consequently offers more complexity in assembly and propellant line hook-ups. The common stage concepts require docking the forward end of stage 1 to the aft end of stage 2 with the major problem being the presence of the main engines. In addition, both stages are operating simultaneously during the time period when stage 1 is returning to Earth orbit and stage 2 is deliverying the payloads. Both nuclear stages emit radiation which can confine the zone of operation around the stages; disposal of the high density reactors presents a more difficult task than for chemical stage components. In addition, the nuclear electric stage must employ a high temperature radiator that is of such size that considerable assembly is required particularly in terms of fluid line hook-ups. The solar electric system also requires on-orbit assembly in terms of the collectors and radiators and has a shorter operating life (in terms of number of missions) than the chemical systems.

Chemical systems are judged to have relatively low technical risks since they are using technologies currently available. Moderate risk was assigned to the nuclear LH_2 system; there has been considerable study and ground test work but no flight experience. Both electric systems have been given high risks since both require development of their basic propulsion systems (MPD thrusters) and also require considerable orbital assembly, yet to be demonstrated.

Masses of LO_2/LH_2 OTV stages are shown in Figure 28 for the various mission options, for the three staging options, indicating potential commonality and the likelihood that vehicle evolution options can be found that will not result in severe

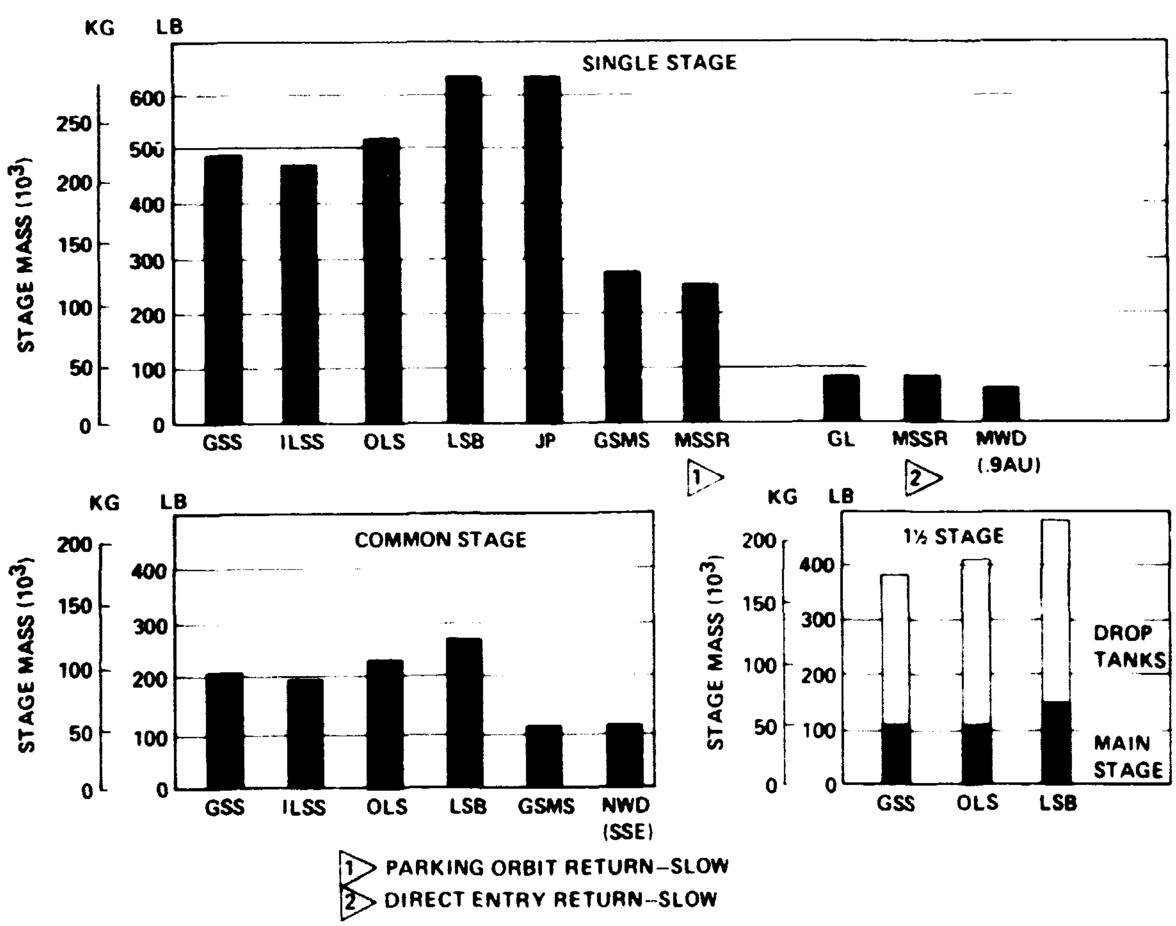


Figure 28. LO_2/LH_2 OTV Applicability

mission/vehicle mismatch penalties. These evolutions could, for example, include use of a common stage second stage in a single stage mode for a smaller requirement.

Lunar Transport Vehicles—Five basic transportation options were considered for the LTV application and are compared in Figure 29. The particular size indicated for each option is related to the payload requirements of the OLS mission; however, the general arrangement will remain the same for all of the missions employing LTV's. It should also be noted that the crew/equipment module even though a payload, is included in each concept since it is never detached.

The illustrated concepts have been considered for the following reasons: The small diameter single stage LO_2/LH_2 concept offers simplicity and reasonable performance (the landing configuration is quite tall if shuttle compatibility is attempted). As a means of reducing the landing height of a single stage LO_2/LH_2 option, consideration is given to a large diameter LTV concept that uses the full diameter capability of the HLLV. Performance gain and improved landing height are possible with a 1 1/2 stage LO_2/LH_2 concept. The drop tank in this case is jettisoned prior to the terminal descent maneuver. Stages employing LO_2/MMH propellant were considered because the average bulk density is approximately three times that of LO_2/LH_2. This factor more than compensates for the lower Isp in terms of the resulting height of the configuration when compared to a LO_2/LH_2 concept using the same staging method.

The 1 1/2 stage LO_2/LH_2 system shown in Figure 30 is used to illustrate the level of detail used in defining LTV configurations.

This LTV concept consists of a main stage attached below the crew/equipment module to form the landing vehicle and an expendable drop tank that is attached above the crew equipment module. At landing, the configuration has a height of 14.6 m (48 ft) and a c.g. 7.1 m (23.3 ft) above the lunar surface.

The drop tank consists of a suspended ellipsoidal LO_2 tank and a LH_2 tank with integral sidewalls. The drop tank is 4.27 m (14 ft) in diameter and 9.8 m (32 ft) in length. The tanks are sized to contain enough propellant to replenish the main stage tanks while the vehicle is in lunar orbit. Propellant remaining in the drop tank is used to provide the deorbit and initial braking delta V of 1642 m/sec (5387 fps). At completion of this maneuver, the tank is jettisoned.

The main stage consists of two cylindrical LO_2 and two LH_2 tanks enclosed within an octagon type body shell. The diameter of the shell is reduced to

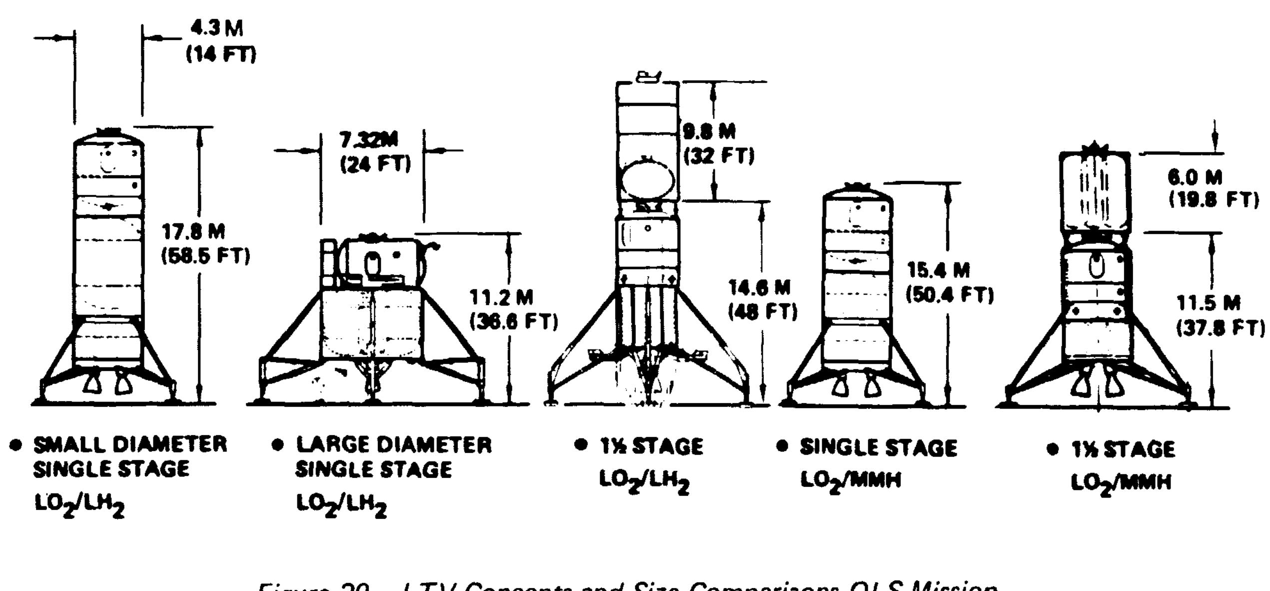


Figure 29. LTV Concepts and Size Comparisons OLS Mission

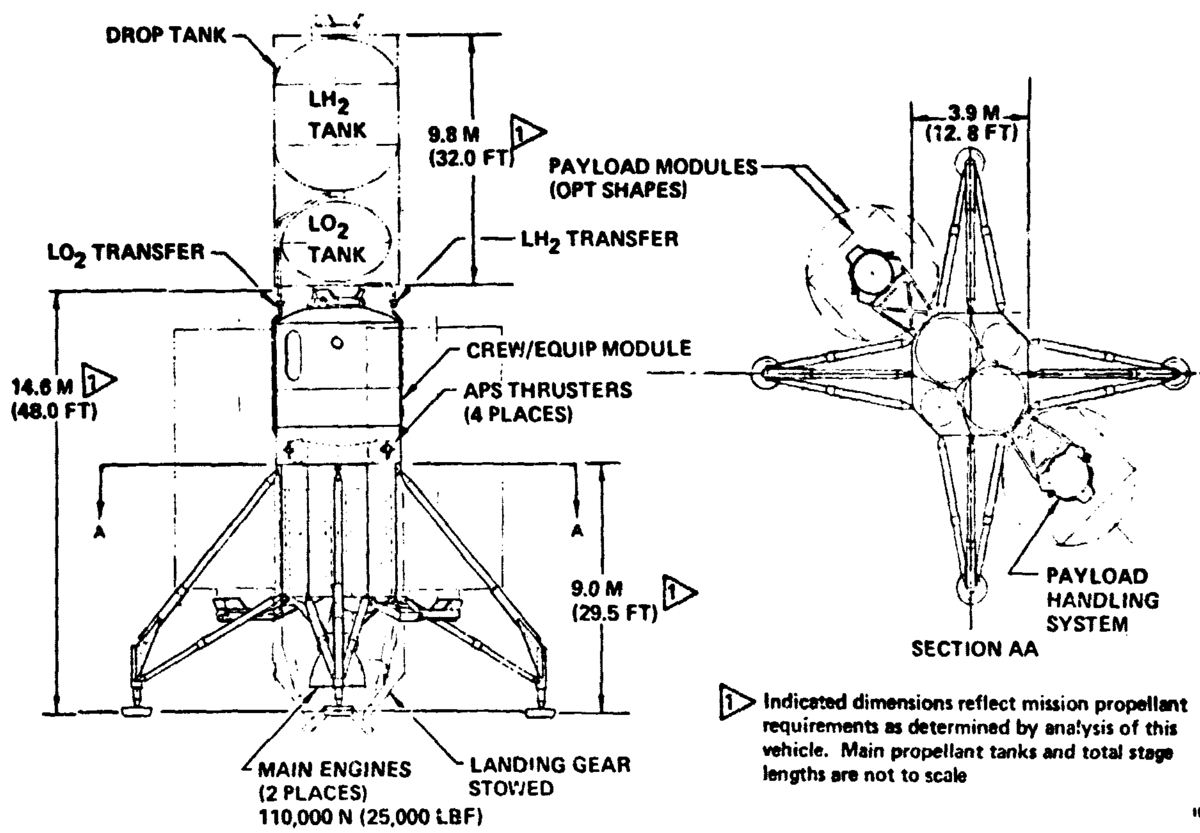


Figure 30. 1-1/2 Stage LO_2/LH_2 LTV Configuration, OLS Mission

3.9 m (12.8 ft) in order to accommodate the stowed landing gear within the 4.27 m (14 ft) physical diameter constraint. The tanks are sized to contain the propellant required to complete the landing maneuver and return the vehicle back to the OLS on lunar orbit.

Electrical power, avionics and APS are located in the bay between the main stage and the crew/equipment module. Thruster clusters associated with the APS are mounted on the side of the equipment module. Payload handling capability is provided by two units attached at the base of the stage. The units are folded down when the vehicle is in the Shuttle. Each unit employs an international type docking mechanism that can be rotated to facilitate docking of payload modules in orbit and also lower them to the lunar surface. Two main stage engines allow for engine out capability with each providing a thrust of 110 000 N (25,000 lb_f). The engines are throttleable and at startburn provide a T/W = 0.3. The landing gear configuration is designed to provide a landing gear radius to c.g. height ratio of 1.2 which is approximately that provided by lunar module.

Mass comparisons of the candidate LTV systems are presented in Figure 31 for the crew rotation/resupply flight which has a delivery requirement of 15 900 kg (35,000 lb) and return of 11 400 kg (25,000 lb).

The 1 1/2 stage LO_2/LH_2 system provides only a slight advantage over the small diameter single stage LO_2/LH_2 concept for the following reasons: 1) the total delta V is low enough that a significant benefit cannot be achieved by staging as is the case when high delta V's are required, 2) the 1 1/2 main stage must employ a shell to support the propellant tanks and 3) the drop tank has been sized to also contain the main stage propellant during transit from Earth orbit.

The greater mass of the large diameter single stage LO_2/LH_2 systems relative to the small diameter concept is due to the additional structural mass associated with the shell that supports the propellant tanks.

The LO_2/MMH stages result in greater mass primarily as a result of their lower Isp (362 vs 453 sec).

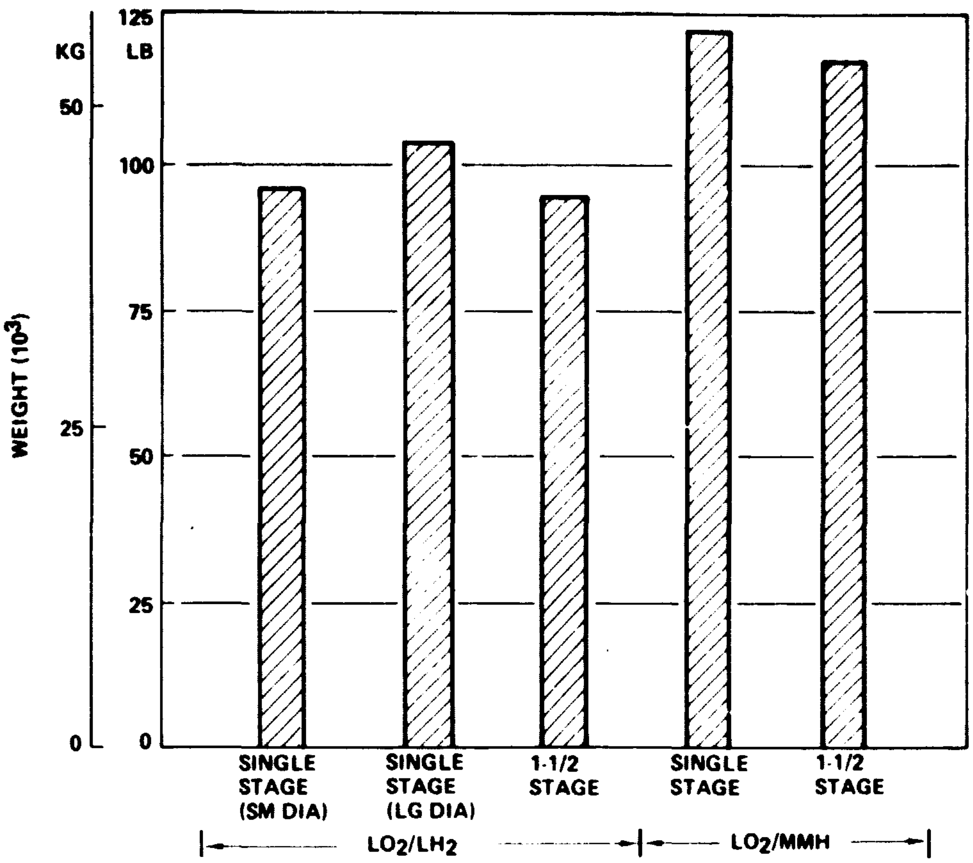


Figure 31. LTV Mass Comparison, OLS Mission

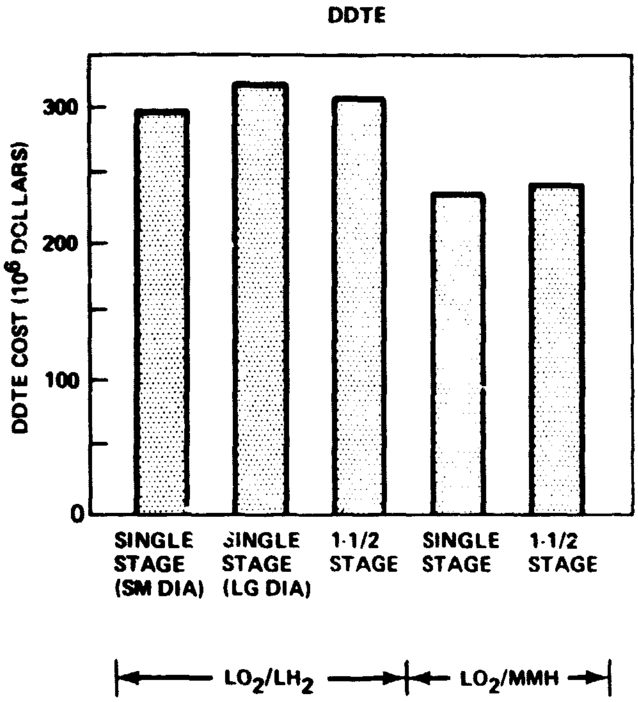
Cost summaries identifying DDTE and first unit cost for the LTV concepts were developed to the same level as previously indicated for the OTV's. A comparison of the total DDTE and first unit cost for the LTV options is shown in Figure 32. Cost values shown do not include the crew and equipment module.

In general, for both the DDTE and first unit cost the LO_2/MMH concepts are cheaper as a result of their main engines being less costly than LO_2/LH_2 engines.

Total propulsion module cost for the LTV options as a function of the number of LTV flights was developed using the DDTE and unit cost from the preceding figure and a $100 per pound launch cost to cover delivery of the LTV hardware and propellant and OTV propellant required for the transfer between Earth and lunar orbits. Results are shown in Figure 33.

Although the MMH concepts begin with the lowest cost, the penalty of their greater propellant weight reflected in terms of transportation cost from Earth to lunar orbit tends to equalize their costs after about ten uses. Therefore, the lowest cost system after approximately 10 LTV flights will be the 1 1/2 stage and small diameter LO_2/LH_2 systems because of their lower replenishment requirement.

Other factors can also be used to assess the various LTV concepts. In terms of full reusability, all of the single stage concepts qualify as do the main stages of the 1 1/2 stage options. Compatibility with the shuttle is primarily concerned with length, diameter and c.g. characteristics. In the case of the small diameter single stage LO_2/LH_2 system the length of the vehicle nearly exceeds the cargo bay limit when the landing gear is stowed. The situation exists when the complete 1 1/2 stage LO_2/MMH system is launched with the shuttle.



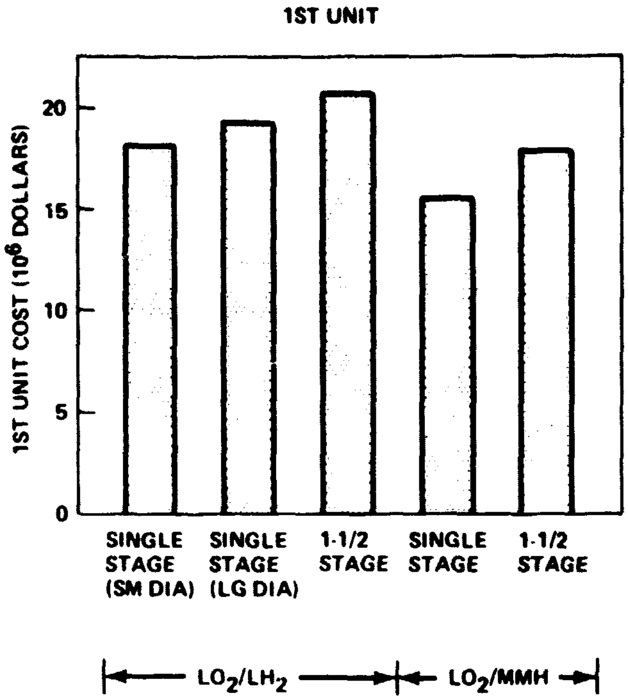


Figure 32. LTV Cost Comparison, OLS Mission

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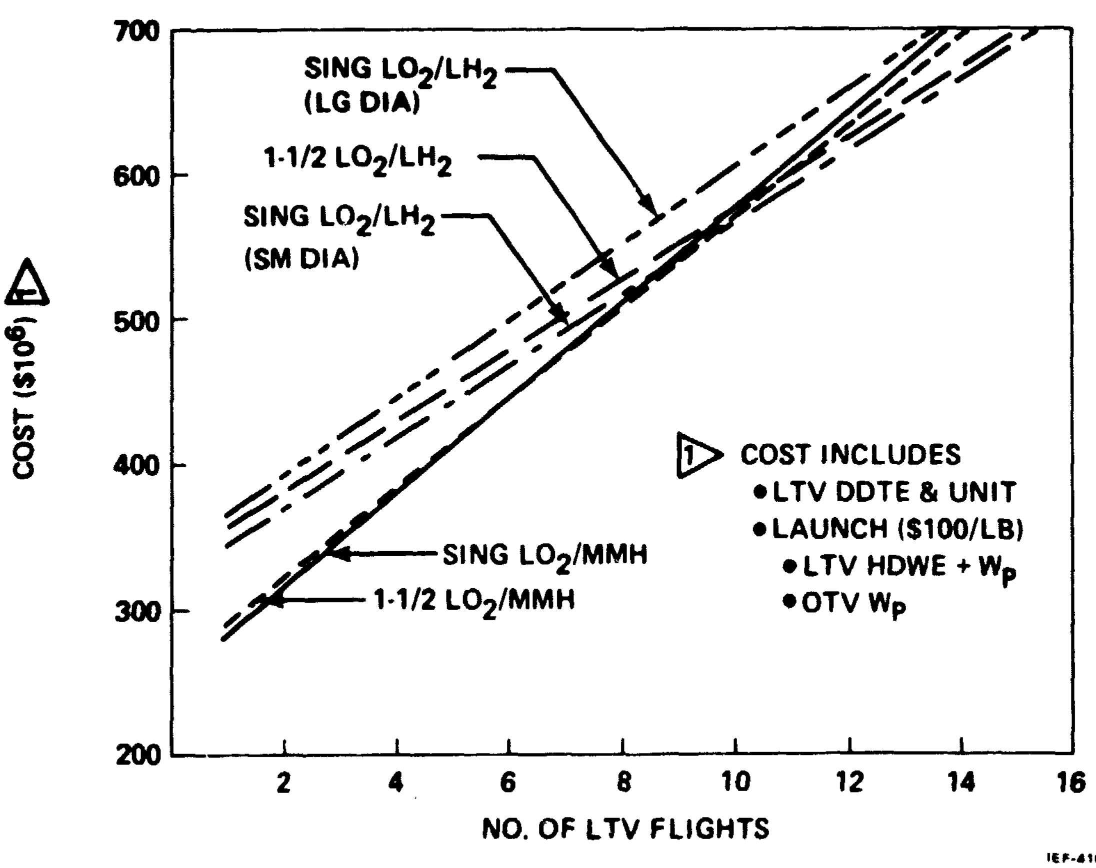

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Figure 33. LTV Cost Sensitivity to Number of Flights

The major concern for the single stage LO_2/LH_2 is that of landing stability even though the specified ratio of landing gear radius/c.g. height has been used. Concern with the height of crewmen above the surface or accessibility is not considered a severe penalty since in all cases an automated elevator type device can be used. The only concern with the 1 1/2 stage concepts is that of drop tank jettisoning and assurance that no impact would occur with the main stage since the system is operating in a "g" field. Propellant line hook-up between drop tank and main stage is considered no more difficult than when a single stage is being refueled in orbit.

3.4 POWER SATELLITE TRANSPORTATION

Selection of a transportation option for the operational phase of the power satellite mission involves several interrelated issues:

- The transportation system will be strongly influenced by selection of assembly location; that selection may be based primarily on satellite-related matters other than transportation.

- Transfer time is important to fleet size determination, to the duration of exposure of the satellite to intense Van Allen radiation, and to delay costs in terms of beginning operation of the satellites.

- Mass to be transported and the cost of transportation are important because the mass is great and the cost must be kept as low as is practicable to maximize economic benefits of the system.

- The selected system must be conducive to that amount of pilot scale demonstration deemed necessary to provide the technical and cost confidence prerequisite to an operational development decision.

These factors are evaluated in the context of a representative power satellite program deploying one operational 10,000 MW satellite per year. The satellite mass will be in the range of 40×10^6 to 100×10^6 kg (90×10^6 to 220×10^6 lb), and will require hundreds of HLLV flights per year.

3.4.1 Assembly and Transfer Options

The choice of assembly and transfer option imposes varying degrees of constraint on orbit transfer transportation. If payloads are to be delivered all the way to geosynchronous orbit before unpackaging and deployment, there are no practical limits on OTV acceleration, but the HLLV-OTV match must be on the basis of simple increments, i.e., the OTV payload is one-half, one, two, etc. HLLV payloads.

If some assembly is to take place in low orbit, the OTV/HLLV match is uncoupled, but OTV accelerations will be limited, depending on degree of assembly and deployment, to relatively low values by satellite element structural limits.

In the low orbit assembly option, accelerations will be limited to very low values in the electric propulsion range. Assembly of an entire power satellite in low orbit appears to be precluded by gravity gradient loads and air drag; four to ten large modules, that can be joined in geosynchronous orbit by relatively simple operations, are indicated. The large size of these modules will probably dictate the use of several OTV's operating in a tuglike fashion, since a single very large OTV would itself require orbital assembly.

Some assembly location factors are not directly associated with transportation of the satellite itself. Crew logistics and other crew considerations strongly favor low orbit assembly; impact on the power satellite itself favors high orbit assembly. Present estimates are that considered all together, these factors tend to favor low orbit assembly.

The most notable difference between high thrust and low thrust systems is trip time. A high thrust vehicle can make a geosynchronous round trip in one day. A low thrust vehicle may require six months. This in turn affects fleet size and fleet investment. For a high thrust reusable system, even if hundreds of trips per year are required, only a few vehicles may suffice since one vehicle can probably make 50 trips per year. For low thrust systems, one vehicle can make perhaps two trips per year and a very large fleet is needed. The self-power concept uses power output from the satellite modules to drive electric propulsion systems for orbit transfer. Typically six propulsion modules per satellite module are required. For completion of one satellite per year, twelve propulsion modules, plus spares, are required.

3.4.2 Transfer System Options

Transfer system options investigated include LO_2/ LH_2 and Nuclear LH_2 (Nerva) high-thrust vehicles, nuclear and solar electric low-thrust vehicles, and self-power, using the satellite module electric output to drive the ascent propulsion system.

Shown in Figure 34 is a representative self-power concept. The satellite consists of four power generation modules (PGM's) and a power transmission system (antenna). The power generation modules are assembled in a low orbit and fly independently to geosynchronous orbit; one of the four must deliver the antenna. In geosynchronous orbit, the four modules are joined to form the complete system. For transfer a propulsion module is attached to each vertex of the PGM concentrator and an additional one may be attached at the thermal cavity, depending on control requirements. Each propulsion module consists of electric and chemical (LO_2/LH_2) thrusters, power conditioning and control equipment, thermal control radiators, and LO_2/LH_2 tankage for return propellants. Ascent propellants are fed from tanks permanently attached to the PGM. During ascent, electrical thrust is used except when the vehicle is shadowed by the Earth; then chemical thrust is used to maintain the PGM in a sun-facing attitude. During the initial part of the transfer, a significant proportion of total thrust is expended to maintain PGM attitude control in the presence of gravity gradient disturbing torque. The penalty for this has been roughly estimated as a 10% addition to total ascent delta V requirement; this has been reflected in the ascent delta V requirement of 6600 m/sec (19,685 ft/sec) used to size the systems. The propulsion modules return to low orbit for reuse using their LO_2/LH_2 thrusters and internal propellant.

3.4.3 Comparison of Options

The mass and cost penalties associated with orbit transfer can be expressed as burden factors to simplify expression of total transportation costs. Shown in Figure 35 are mass burdens, expressed as (orbit transfer mass increment in low orbit)/(power satellite mass). The mass burden for low thrust systems is dependent on the performance of the power generation system in terms of mass/power ratio, if trip time is held constant. A 160-day round trip was groundruled, offering the potential for two trips per year per vehicle. For self-power, a 90 day up-trip was assumed since this thrust (i.e.

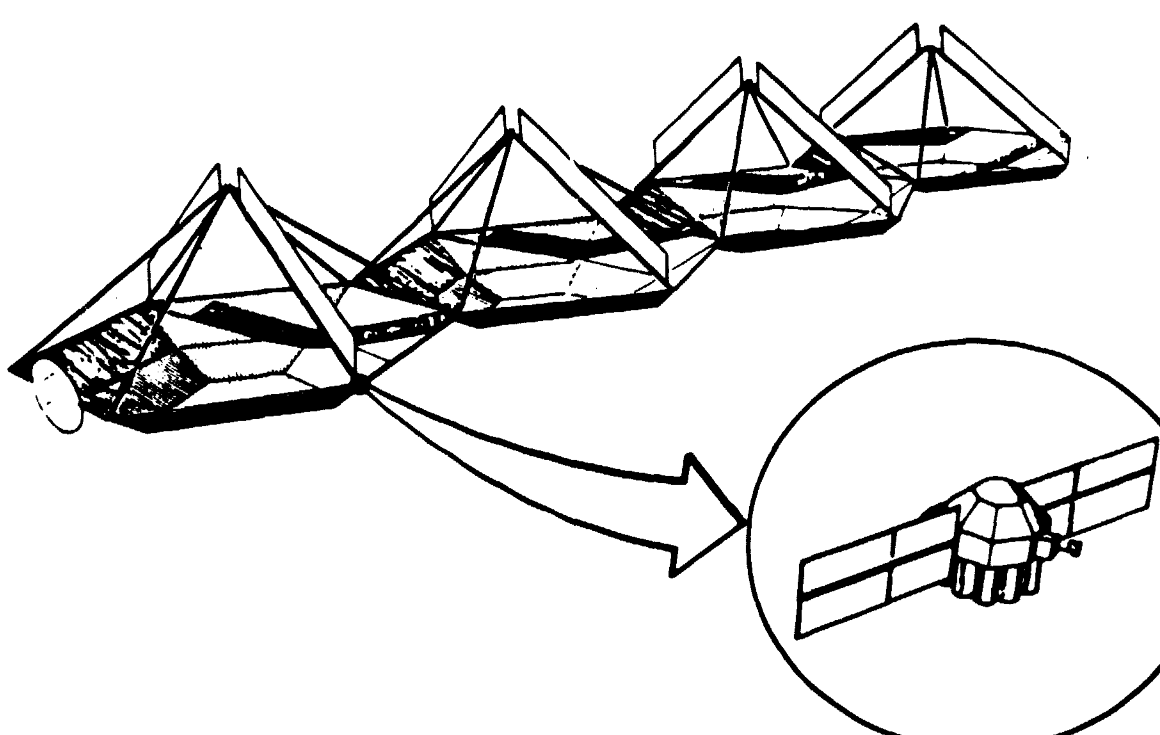

Figure 34. Self-Power Concept

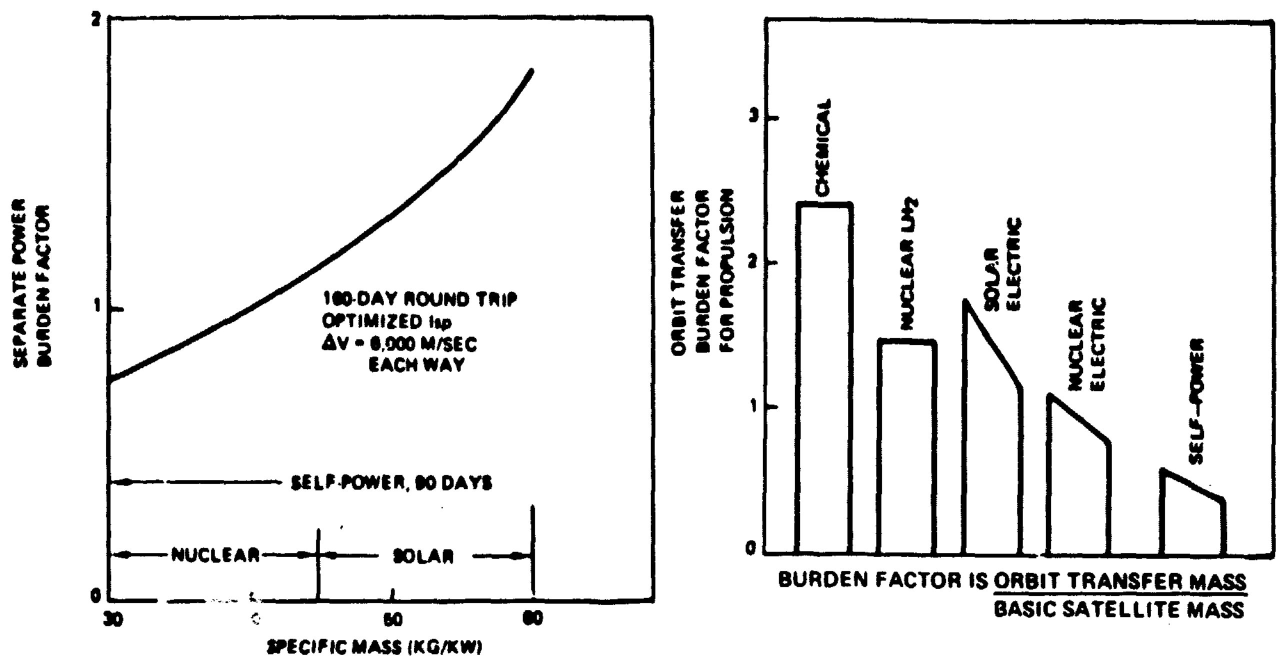

Figure 35. Orbit Transfer Burden Factors

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acceleration) level appears to be compatible with gravity gradient control requirements.

Recurring cost for power satellite orbit transfer can be estimated by considering HLLV costs in combination with mass burden factors plus orbit transfer hardware cost, as indicated by the equation in Figure 36.

Packaging and orbital assembly operations logistics have not been studied in any detail and the values shown are ROM estimates. The packaging burden is (actual HLLV flights required per satellite)/(ideal number considering only satellite total mass). The logistic burden is (cost of all logistics flights to support assembly or one satellite)/(cost of ideal number of HLLV flights). Many of the logistics flights are expected to employ the shuttle. Orbit transfer mass burdens were shown on a previous chart.

Recurring cost results are shown on the right-hand side of the chart. Orbit transfer system DDT&E costs were not included but are nearly negligible at a power satellite activity level. The apparent advantage of self-power results because

- The mass burden is low
- The hardware cost is far less than other systems with low mass burden because power generation is provided by the satellite modules.

This analysis did not include the effect and cost of self-power design requirements on the power satellite module. Therefore, and because the self-power system involves significant technical risk, the other options are still considered viable.

A final very important issue is ability to demonstrate technical capability and potential for low cost in a pilot program. Principal considerations are as follows:

- Low orbit is strongly favored for pilot plant orbital assembly and test (need only Shuttle and space station).

COST = HLLV COST (F_P + F_O + F_T)
+ TRANSFER HARDWARE COST

WHERE

F_P = PACKAGING BURDEN' ESTIMATED 1.1

F_O = ORBITAL LOGISTICS BURDEN, ESTIMATED 0.1

F_T = ORBIT TRANSFER MASS BURDEN

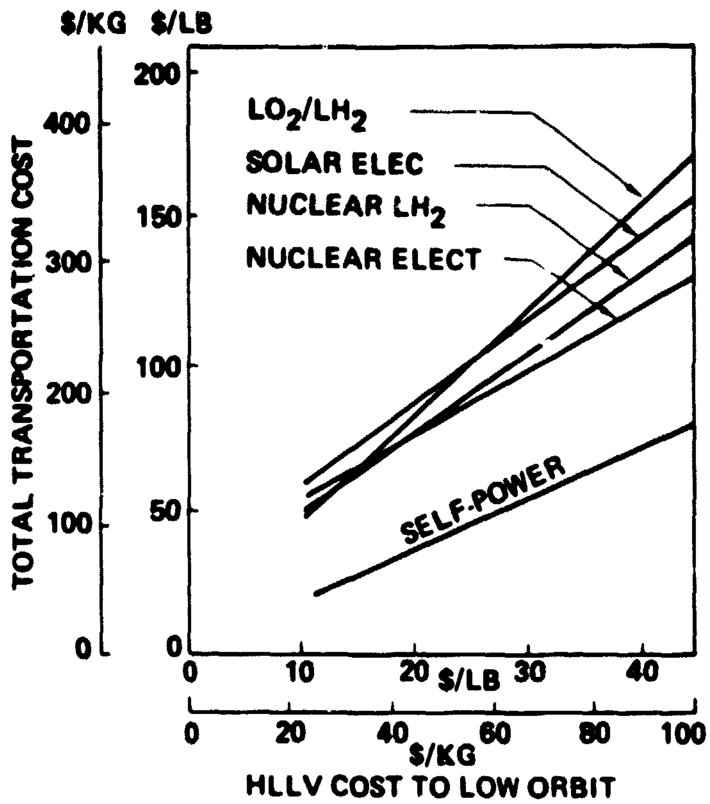

Figure 36. Power Satellite Orbit Transfer Cost Summary

- Pilot plant demonstration is expected to include geosynchronous transfer and further tests.

- Issues are similar to those for operational plant except that OTV DDT&E dominates cost.

A typical pilot plant mass is estimated as 250 000 kg (550,000 lb). Total transportation masses are compared in Figure 37. The OTV options used for this comparison, excepting self-power, were adopted from the geosynchronous space station options described previously. Trip time constraints were relaxed in view of the relatively few trips required.

Transportation cost comparisons are presented in Figure 38, including OTV DDT&E cost. The shuttle is assumed as the Earth launch vehicle. Since a high thrust chemical OTV will be needed for crew transport, an alternate comparison would delete the DDT&E for this system as it is required regardless. On that basis, the LO_2/LH_2 system is minimum cost. Self-power could be demonstrated at small additional cost, trading its DDT&E against reduction in number of shuttle flights.

4.0 FSTSA PHASE II STUDY PLANS

Scenerios (example futures) will provide the working basis for the Phase II effort. Four working scenarios have been developed. Transportation system options will be fitted to each scenario, defining a candidate evolution for each option. Since there will be alternative transportation system options applicable to each scenario, there will be about 10 to 20 candidate programs to be evaluated during the first optimization activity. (A program is a scenario employing a particular transportation evolution; each scenario can employ a number of alternative transportation options.) Program cost data will be developed using the cost data base developed by Phase I. Only transportation and transportation-related costs will be determined; mission hardware will not be costed. Transportation costs will include DDT&E, fleet investment, recurring launch and operations, and sustaining costs.

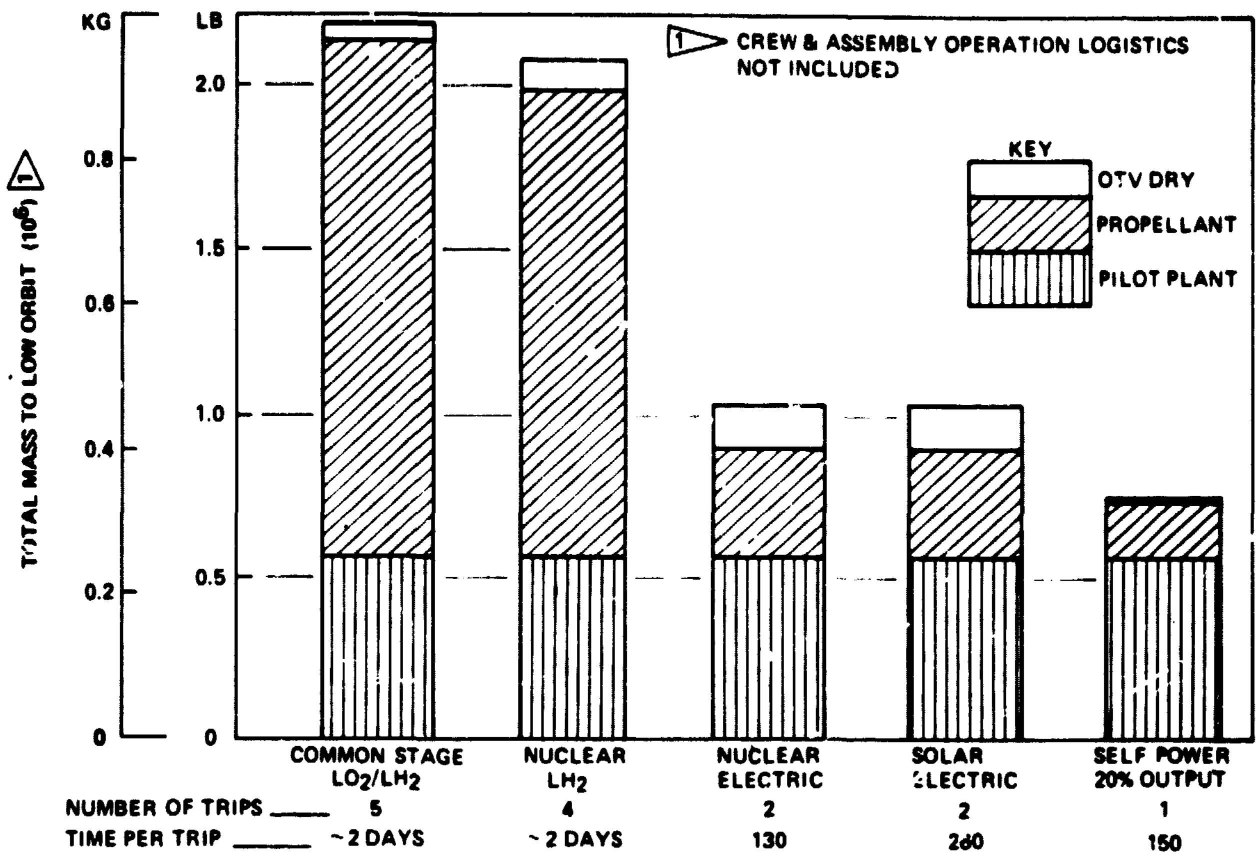

Figure 37. Power Satellite Pilot Plant OTV Options

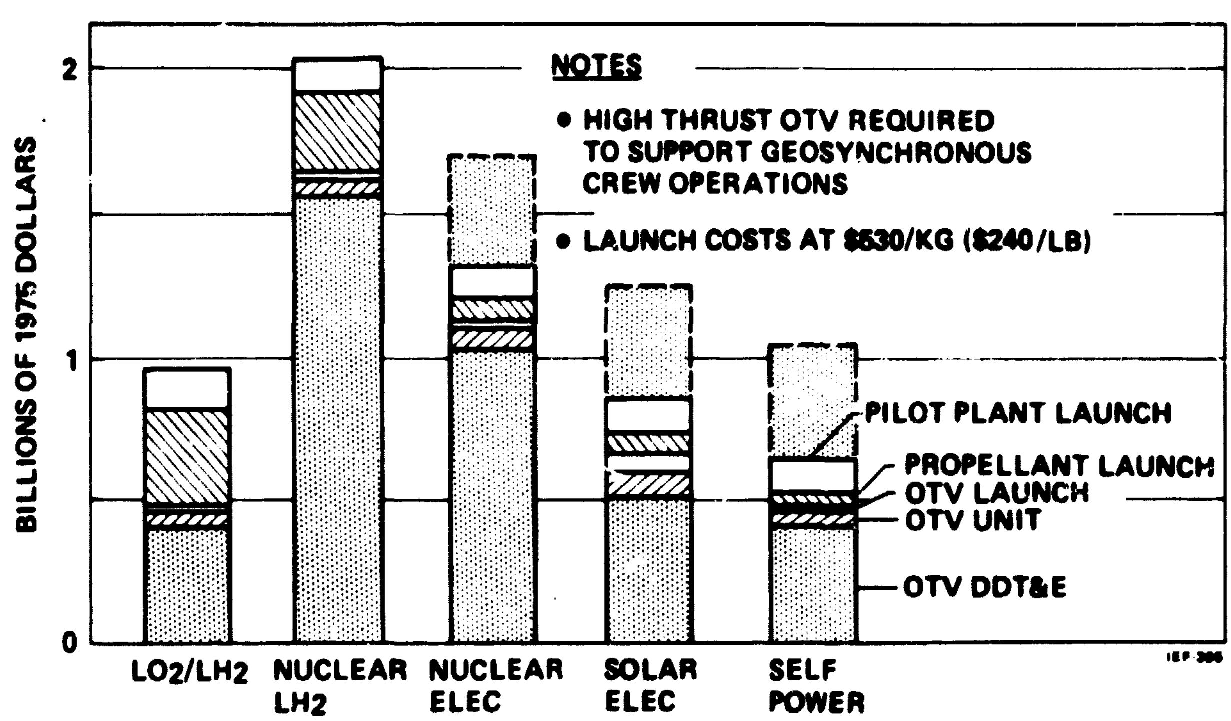

Figure 38. Transportation Cost Comparison Power Satellite Pilot Plant

Optimization and evaluation will be performed as follows:

- Representative programs (scenarios with transportation options) will be selected for modeling. Sections will be made to ensure that key questions are addressed. Parametric models of the program will be constructed, relating such factors as heavy lift launch vehicle (HLLV) payload, HLLV reusability, orbit transfer vehicle (OTV) payload, OTV size, number of flights, and costs.

- Results of the modeling will be evaluated in order to select discrete transportation solutions most nearly corresponding to the idealized optimal points determined by analysis.

- Qualitative evaluations, employing qualitative criteria from the Phase I extension, will also be made.

- Results will be reviewed with NASA at the pre-midterm working session and initial selections made.

- This process will be repeated with refinements during the second half of the study.

Alternative scenarios will be prepared for the purpose of testing evolutionary development strategies. Strategies analysis will proceed through steps of commonality identification, decision driver identification logic tree development and method of application. The resulting evolutions will be evaluated. Results of the tests and evaluations will be used to revise and update the strategies. Also during the second half of the study, transportation system definitions will be developed. These will utilize data developed during the Phase I extension and will describe the systems to a level of detail adequate as a point of departure for any subsequent Phase B definition studies that the NASA may elect to conduct.